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PRE-INJECTION TRAJECTORY CHARACTERISTICS REPORT AC-14

GENERAL DYNAMICS
Convair Division

PRE-INJECTION
TRAJECTORY CHARACTERISTICS
REPORT
AC-14

GDC-BKM67-064
September 1967

Contract NAS3-8711

Prepared by G.D. Petzel
G. D. Petzel

Approved by D.H. Lesney
D. H. Lesney
Research Group
Engineer

Checked by R. L. Stone for
L. G. Wilson
Design Specialist

Approved by F.B. Anthony
F. B. Anthony
Asst. Chief Engineer
Flight Mechanics

GENERAL DYNAMICS
CONVAIR DIVISION
San Diego, California

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FOREWORD

This report has been prepared in response to the requirements of NASA Contract NAS3-8711 as outlined in Item 56 of Reference 7. Preflight trajectory and vehicle characteristics and trajectory design rationale are summarized for the purpose of providing a central information document for the Atlas/Centaur AC-14 flight. All information presented is the best known at the time of publication (10 weeks prior to the prime launch date). Any significant changes will be published at the time of occurrence.

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SUMMARY

This report presents preflight pre-injection trajectory characteristics for the AC-14 flight. In addition it summarizes the AC-14 trajectory design rationale. This document should be considered as the general source for AC-14 trajectory and performance information.

The scope of this report includes the following:

a. Mission
b. Performance
c. Range Safety
d. Tracking
e. Flight Mechanics
f. Trajectory Design Criteria
g. Vehicle Configuration
h. Propulsion
i. Vehicle Systems
j. Weights
k. Aerodynamics
l. Physical and Atmospheric Data

Data contained in this report are the best available at the time of publication. Slight variations in some of the nominal reference vehicle data will necessarily occur due to last minute weight changes, wind conditions, et cetera. The data given here however are expected to be representative of the AC-14 flight. All times and dates presented in this report are based on Greenwich Mean Time (GMT).

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INTRODUCTION

The intent of this report is threefold: 1) to present pre-injection trajectory characteristics for the AC-14 launch vehicle, 2) describe trajectory design rationale, and 3) demonstrate that the trajectory is within allowable design limits.

AC-14 is the fourteenth Atlas/Centaur launch vehicle scheduled for flight. For the sixth time a Surveyor spacecraft (A-21 series) will be injected into an earth-to-moon transfer orbit. Lunar encounter will occur approximately 66 hours after injection. The basic supporting data and criteria that have led to the final AC-14 trajectory design are presented in the following sections.

SECTION 1

MISSION

The AC-14 vehicle will be launched from Cape Kennedy with a flight profile designed to inject a Surveyor spacecraft into an earth-to-moon transfer orbit. Lunar encounter will occur approximately 66 hours after injection. A parking orbit mode of ascent will be flown, i.e., two Centaur main engine powered phases interrupted by a coast phase. Flight profile design will encompass launch-on-time features required to compensate for changing earth-moon geometry in the event of launch delays.

A summary of major parameter constraints and requirements is given in Table 1-1. Parameter ranges indicated are operating limits. The actual parameter values will be launch-day/launch-time dependent.

Table 1-1. Mission Parameter Summary

Primary Launch Period	11/5/67 through 11/12/67 (GMT)
Launch Mode	Parking Orbit Ascent
ETR Launch Site	Complex 36B
Launch Azimuth Sector[1]	78 through 115 deg[2]
Parking Orbit Coast Time[1]	116 sec[3] (minimum) through 25 min[4] (maximum)
Parking Orbit Altitude	90 n.mi.
Injection True Anomaly (MECO 2)	≈ 4 deg
Injection Energy[1]	-1.70 through -0.85 km^2/sec^2[5]
Flight Time	≈ 66 hr

[1] Launch day dependent.
[2] Current limits as approved by ETR range safety.
[3] Minimum ullage settling time.
[4] Design limit.
[5] Limits used for parametric trajectory studies.

MISSION PROFILE

The powered flight phase has been shaped to maximize payload capability within the specified mission, configuration, and trajectory constraints. The key features of trajectory optimization and design are presented in Section 6 of this report. Figures 1-1 and 1-2 illustrate the powered-ascent flight profile and the overall in-plane trajectory profile for lunar encounter. Figure 1-3 depicts the earth ground-trace envelope, during the initial 15 hours of flight, for the 78- to 115-degree launch azimuth sector.

The Atlas/Centaur configuration rises vertically from liftoff (defined as 2-inch motion above the launch pad) until 15 seconds of flight time has elapsed. During the last 13 seconds of this interval the Atlas-stage flight control system rolls the vehicle from the launcher-aligned azimuth to the desired flight azimuth. The vehicle then executes a preprogrammed pitch maneuver in the downrange direction. Termination of the booster-phase flight is initiated by a staging discrete (BECO) issued by Centaur guidance when an acceleration level of 5.7 g's is sensed. A backup staging-command signal is provided by an Atlas-stage accelerometer at a preset level of 5.90 g's. The booster package is jettisoned 3.1 seconds after the staging discrete is issued.

Centaur-guidance steering signals are admitted to the Atlas-stage autopilot eight seconds following BECO, and the system operates in a closed-loop mode throughout the remainder of the flight. During the sustainer-phase flight, insulation panels and nose fairing are jettisoned. The sustainer phase is terminated by a discrete (SECO) from a pressure sensor in the fuel manifold in response to oxidizer depletion and causes the sustainer and vernier engines to be shut down. A short time later (2 seconds) the Atlas stage programmer energizes the electrical disconnect, fires the flexible linear-shaped charge to separate the Centaur stage, and fires the eight Atlas retrorockets to back the Atlas away from the Centaur.

Prior to SECO the Atlas programmer initiates the Centaur-stage prestart sequence. The boost pumps are started and brought up to speed. Propellants flow through the

Centaur fuel and oxidizer system, chilling down the hardware to preclude cavitation at Centaur first main engine start (MES 1).

The signal for starting the Centaur main engines is issued by the guidance programmer. Guidance steering commands are nulled at SECO and readmitted at MES 1 + 4 seconds, after the engine start transient has passed. Centaur first main engine cutoff (MECO 1) is commanded by the guidance system when the required circular orbit insertion conditions for a 90-nautical mile parking orbit are almost achieved.

Subsequent to MECO 1, two 50-pound thrust rockets provide initial propellant settling and circularization of the parking orbit. After the initial propellant settling phase is completed a set of two 3-pound rockets provides a continuous propellant level control throughout the parking orbit coast phase until initiation of the prestart events for Centaur second main engine burn. The vehicle attitude during parking orbit coast is aligned with the inertial velocity vector via steering commands issued by the guidance system.

Centaur second main engine start (MES 2) is preceded by operation of the two 50-pound rockets to ensure positive propellant settling. During the propellant settling phase the boost pumps are restarted and the main engines again chilled down. Guidance steering commands are nulled during the first four seconds of main engine operation and then readmitted. The Centaur second main engine cutoff (MECO 2) is commanded by the guidance system when the required injection conditions are achieved.

Subsequent to termination of Centaur second powered phase, the programmer provides timed discretes for: (1) separating the spacecraft from Centaur, (2) reorienting Centaur 180 degrees with respect to the velocity vector at MECO 2, and (3) venting propellants through the engine thrust chambers and vent valve for retrothrust.

A detailed sequence of major flight events and associated times is presented in Section 1.2.

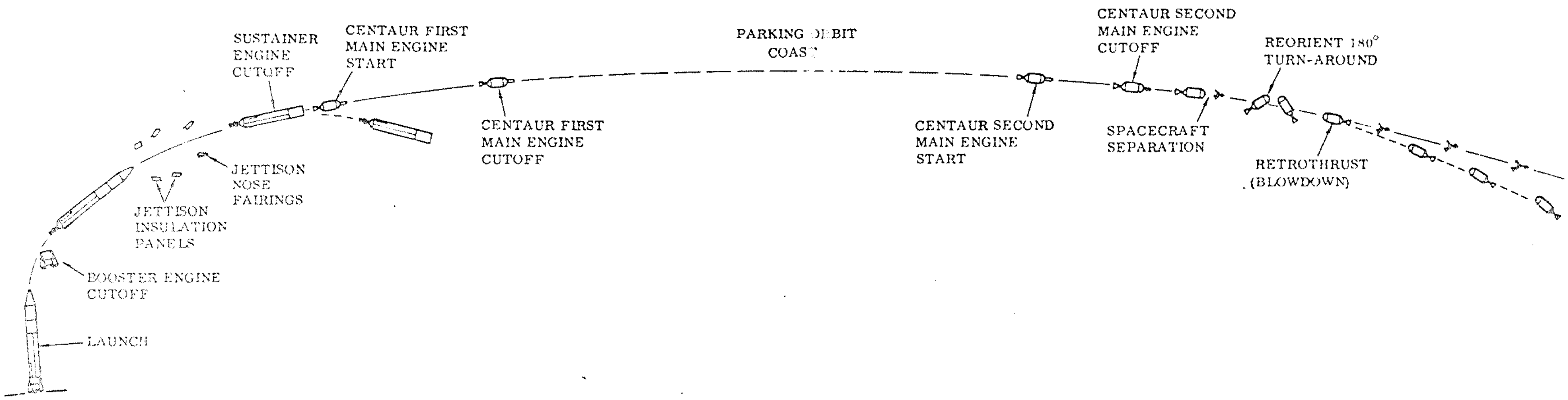

Figure 1-1. Powered Ascent Flight Profile

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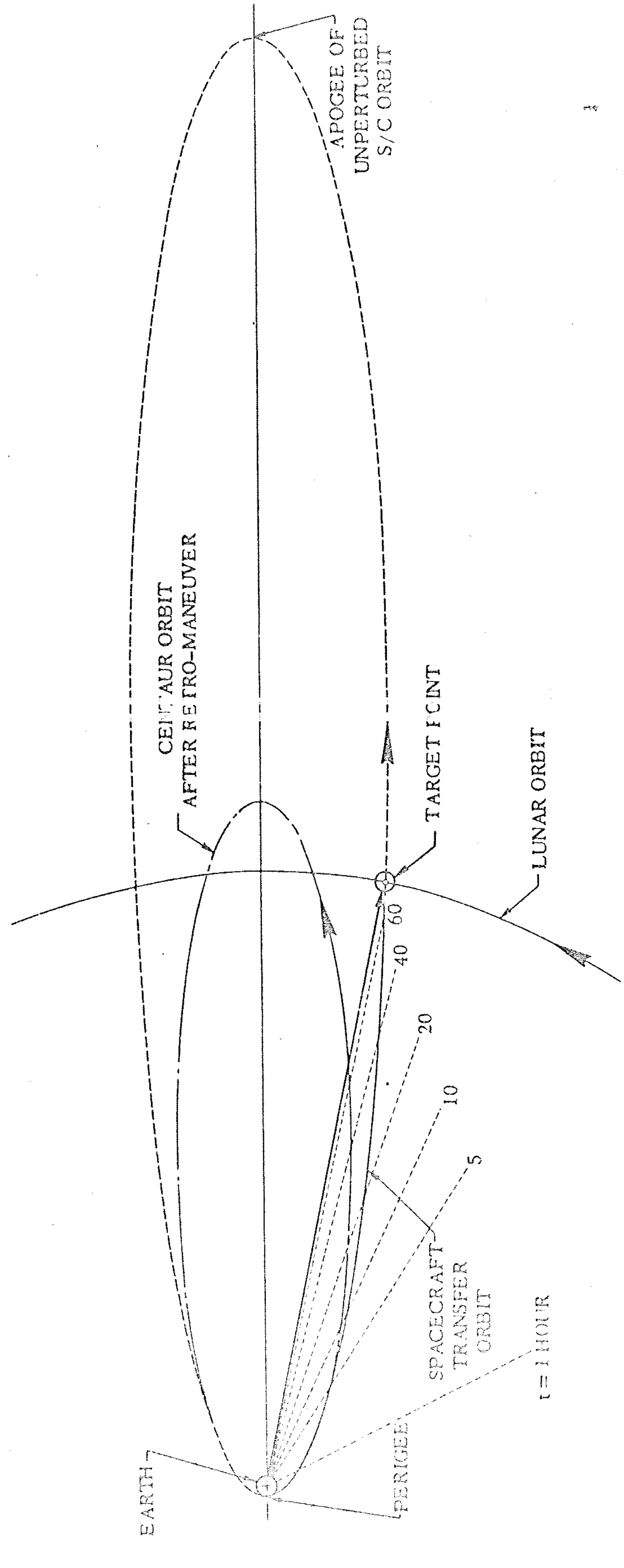


Figure 1-2. Typical Lunar Transfer Orbit

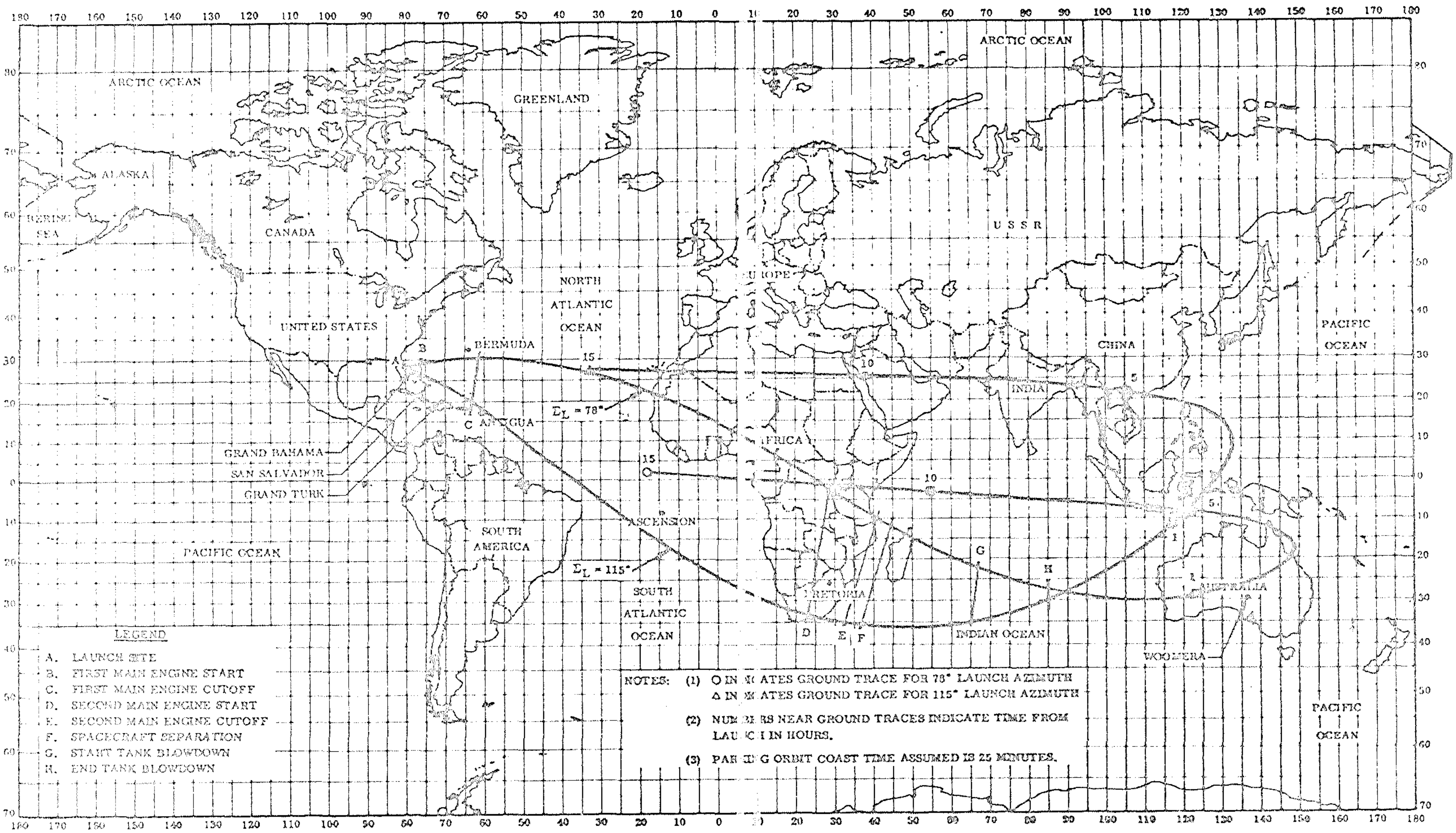


Figure 1-3. Typical Spacecraft Trajectory Earth Ground Trace Envelope

1.2 SEQUENCE OF EVENTS

The sequence of events for the AC-14 ascent trajectory and assocated nominal times, based on References 1 and 4, are listed in Table 1-2. Event times are presented in order of occurrence and are referenced to launch vehicle 2-inch motion (liftoff). Nominal times are based on a trajectory simulation targeted for a launch date of 11/7/67 and azimuth equal to 78 degrees. Nominal times for other azimuths and launch dates are within 0.01 second of Table 1-2 values through Centaur main engine start. MECO 1 and on nominal times vary from trajectory to trajectory due to performance changes resulting from launch azimuth and transfer orbit energy variations with launch time.

The nominal MECO 1 discrete can occur between B + 329.4 and B + 331.8 seconds (reference 2). After adding a 3 σ uncertainty of -7.5 and +8.5 seconds (reference 28) the MECO 1 discrete can now occur between B + 321.9 and B + 340.3 seconds. The Enable-MECO and MECO-Backup discretes are given at B + 300.0 and B + 356.5 seconds respectively (reference 1). From Figure 1-4 it can be seen that the MECO 1 discrete is within the allowable range.

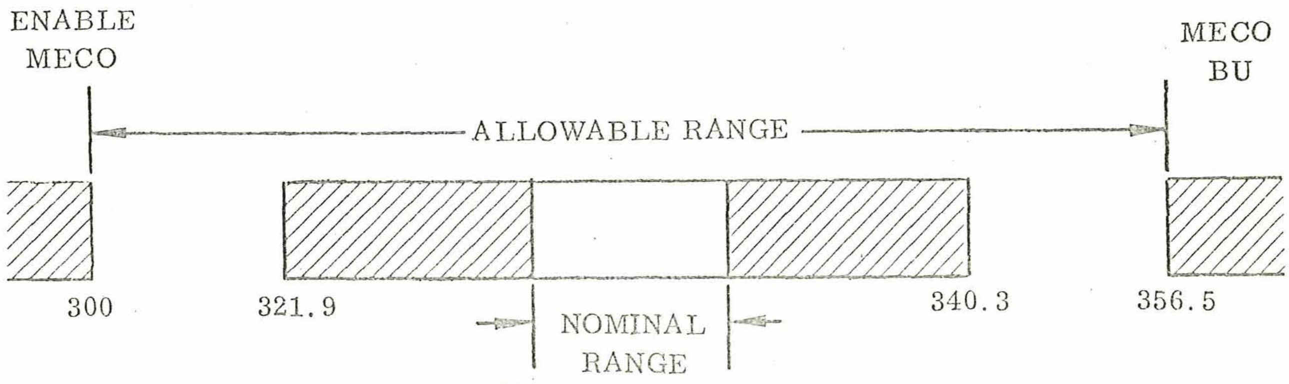


Figure 1-4. MECO 1 Compatibility

The MECO 2 discrete can occur nominally between D+172.2 and D+173.6 seconds (reference 2). With a 3σ uncertainty of ± 2.7 seconds (reference 28) the MECO 2 discrete can now occur between D+169.5 and D+176.3 seconds. The Enable-MECO and MECO-Backup discretes are given at D+143 and D+192 seconds respectively (reference 1). From Figure 1-5 it can be seen that the MECO 2 discrete is within the allowable range.

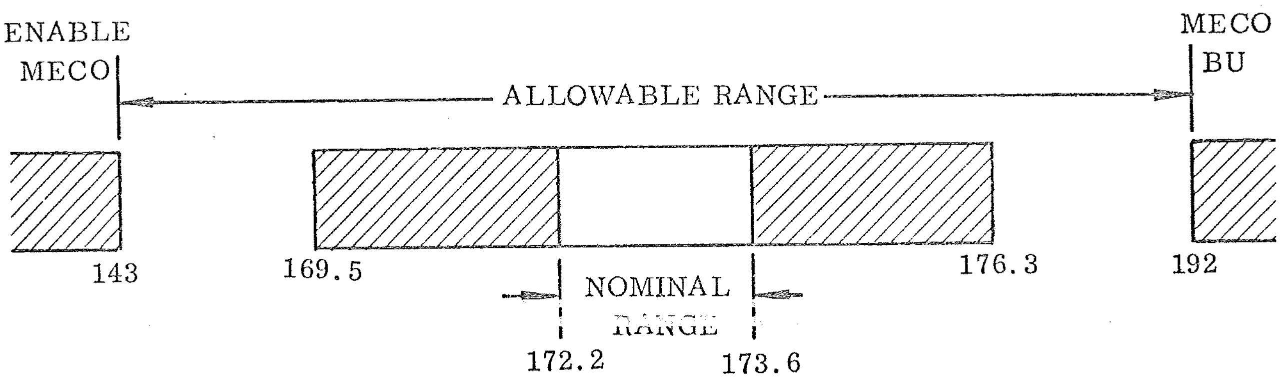


Figure 1-5. MECO 2 Compatibility

The uncertainty range of BECO is ± 2.6 seconds which establishes its lower limit at 151.1 seconds. Enable Staging occurs at 148 seconds which puts BECO within the allowable range.

Table 1-2. Sequence of Events[1]

EVENT	TIME (sec)	TIME (sec)
Two-Inch Motion (Liftoff)	A + 0	T + 0
Start Roll	A + 2	T + 2
End Roll	A + 15	T + 15
Start Pitchover	A + 15	T + 15
Enable Booster Staging	A + 148	T + 148
*Booster Engine Cutoff (Guidance Discrete, Staging Acceleration 5.7 g's)	B + 0	T + 153.7
*Jettison Booster Package	B + 3.1	T + 156.8
Admit Guidance Steering	B + 8	T + 161.7
*Jettison Insulation Panels	B + 45	T + 198.7
Start Centaur Boost Pumps	B + 62	T + 215.7
*Jettison Nose Fairing	B + 75	T + 228.7
Enable Atlas/Centaur Separation	B + 84	T + 237.7
*Sustainer Engine Cutoff (SECO, by Propellant Depletion)	C + 0	T + 248.8
Inhibit Guidance	C + 0	T + 248.8
*Atlas/Centaur Separation	C + 1.9	T + 250.7
Fire Atlas Retrorockets	C + 2	T + 250.8
First Prestart (Centaur Engine Chilldown, MES 1-8 sec)	C + 3.5	T + 252.3
*First Centaur Main Engine Start (MES 1)	C + 11.5	T + 260.3
Admit Guidance	C + 15.5	T + 264.3
Enable First Centaur Main Engine Cutoff	C + 300	T + 548.8
*First Centaur Main Engine Cutoff (MECO 1, Guidance Discrete)	C + 329.9	T + 578.7
Start Ullage Motors (V[2])	C + 329.9	T + 578.7

*Asterisk denotes a MARK event.

[1] Parking Orbit Coast Time is 15.2 minutes.

[2] See Section 8.2.2 for Engine Designation.

Table 1-2. Sequence of Events[1] (Contd)

EVENT	TIME (sec)	TIME (sec)
First Centaur Main Engine Cutoff Backup (MECO 1 BU)	C + 356.5	T + 605.3
*Stop Ullage Motors (V^2) and Start Propellant Settling Motors (S^2)	C + 407.4	T + 656.2
Guidance Discrete (L 1)	D + 0	T +1429.7
*Stop Propellant Settling Motors (S^2) and Start Ullage Motors (V^2, MES 2-40 sec)	D + 20	T +1449.7
Start Centaur Boost Pumps (MES 2-28 sec)	D + 32	T +1461.7
Second Prestart (Centaur Engine Chilldown, MES 2-17 sec)	D + 43	T +1472.7
*Second Centaur Main Engine Start (MES 2)	D + 60	T +1489.7
Stop Ullage Motors (V^2)	D + 60	T +1489.7
Inhibit Guidance	D + 60	T +1489.7
Admit Guidance	D + 64	T +1493.7
Enable Second Centaur Main Engine Cutoff	D + 143	T +1572.7
*Second Centaur Main Engine Cutoff (MECO 2) and Inhibit Guidance	D + 173.1	T +1602.8
Second Centaur Main Engine Cutoff Backup (MECO 2 BU)	D + 192	T +1621.7
*Extend Surveyor Landing Gear	D + 194	T +1623.7
*Unlock Surveyor Omni Antennas	D + 204.5	T +1634.2
*Turn on Surveyor High Power Transmitter	D + 225	T +1654.7
*Separate Spacecraft Electrical Disconnect	D + 230.5	T +1660.2
*Separate Spacecraft	D + 236	T +1665.7
Admit Guidance	D + 241	T +1670.7
*Start 180 Degree Turnaround	D + 241	T +1670.7
*Start Ullage Motors (V^2)	D + 281	T +1710.7
*Stop Ullage Motors (V^2)	D + 301	T +1730.7
*Start Retrothrust	D + 476	T +1905.7
*Stop Retrothrust	D + 726	T +2155.7
Start Ullage Motors (V^2)	D + 726	T +2155.7
*Energize Power Changeover Switch	D + 826	T +2255.7

1.3 TARGETING

Targeting is the trajectory design procedure used to determine the required navigation for achieving the prescribed target conditions. This procedure involves an iterative process of improving starting values until desired landing conditions are met. For Surveyor missions, the landing conditions are specified to Convair by the Jet Propulsion Laboratory. The specified end conditions (Table 1-4) are: selenocentric latitude, selenocentric longitude, unretarded landing velocity, and earliest lunar arrival times. Three end conditions are satisfied in the targeting procedure. This requires at least three independent injection parameters which influence the lunar landing. The independent targeting variables used are launch time, parking orbit coast time, and injection energy for a given launch azimuth. In the targeting procedure, the first three end conditions are satisfied and then the arrival time constraints are checked. If the targeted arrival time is not equal to or between the specified arrival time constraints, the unretarded landing velocity is allowed to vary from the specified value in order that an arrival time constraint can be met.

The results of targeting are the launch-day-dependent guidance constants, the launch-azimuth/launch-time relationship, and performance and trajectory characteristics. The time-dependent launch parameters obtained from the targeted trajectories are assembled to form "firing tables". Firing tables are used for

a. Establishing the launch-day-dependent trajectory data.

b. Establishing the launch-day-dependent guidance constants.

c. Establishing the launch azimuth settings and the time interval in which these launch azimuth settings can be used.

The day-dependent constants and the azimuth/time relationship are, generally, determined only once because they are not greatly influenced by subsequent launch vehicle configuration changes. Performance and trajectory characteristics, however, must usually be redefined to reflect configuration changes. For detailed information regarding targeting, the reader is referred to the AC-14 Firing Tables report, Reference 2.

1.3.1 GUIDANCE CONSTANTS

The launch-day-dependent guidance constants (the J constants) are used along with launch time to determine the prelaunch guidance parameters used in the guidance inflight equations to generate steering signals. The computation of the guidance parameters takes place during the prelaunch phase after the GO TO FLIGHT MODE signal is issued. Several hours prior to this, the day-dependent guidance constants are stored in the guidance computer, and then read out for verification.

1.3.2 LAUNCH ON TIME

The launch time required for determination of the guidance prelaunch parameters is calculated when the GO TO FLIGHT MODE signal is issued. About two hours prior to the opening of the firing window (Section 1.3.4), a LAUNCH ON TIME (LOT) signal is issued. The elapsed time following this signal is measured and stored by the guidance system. When the GO TO FLIGHT MODE signal is received, J(1) is subtracted from the elapsed time, and the resulting value is used for calculation of the prelaunch parameters.

For the AC-14 mission the CCLS (Computer Controlled Launch Set) will provide the LOT signal. The GMT of Prime LOT and the original J1 are inputs to the CCLS ground computer. When the LOT signal is issued the difference between the current GMT and the GMT of Prime LOT is calculated and the J1 and JSUM constant modified accordingly.

1.3.3 LAUNCH AZIMUTH

A prelaunch task, independent of the guidance system, but launch-time-dependent, is setting the launch azimuth. The azimuth setting determines the orientation of the trajectory plane through the Atlas booster phase. This function is performed by setting the voltage applied to the Atlas roll position gyro torquer. The input voltage causes the gyro output axis to be torqued at a constant rate to a known angle. The autopilot senses an off-null gyro output and commands vehicle roll rates to null the output. This maneuver is performed during the vertical rise immediately following vehicle liftoff (roll maneuver starts at 2 and ends at 15 seconds after liftoff).

1.3.4 FIRING WINDOW

In addition to time-dependent data for guidance and autopilot systems, data for selecting firing windows are needed. For targeting purposes, the firing window is defined as the gross time interval of permissible launch, considering only 1) the established range safety constraints (launch azimuths of 78 to 115 degrees) and 2) basic launch vehicle constraints, i.e., acceptable Atlas/Centaur performance capability (for the specified targeting criteria and spacecraft weight) and parking orbit coast time within the design limits of 116 seconds to 25 minutes.

While the firing windows are established by Convair for the above constraints, launch windows (the time intervals during which the actual launch will be considered) are based on additional range safety, launch vehicle, spacecraft, mission, and range support constraints set by NASA/LeRC, JPL, and Convair.

Data required for selecting the firing windows include the time variation of Centaur excess propellants, the relationship of excess propellants to the probability of sufficient fuel aboard to accomplish the mission, and the launch-azimuth/launch-time functions.

The relationship between Centaur excess propellants, launch azimuth, and parking orbit coast time as a function of launch time is shown in Figure 1-6 for a typical launch window (launch date - 11/7/67). Excess propellants are defined as the total weight of usable fuel and oxidizer remaining at the completion of a standard nondispersed trajectory (a nominal trajectory). A negative value of excess propellants represents a condition where the propellant supply for a nominal performing flight is depleted before the terminal conditions required to perform the intended mission are reached. Figure 1-7 presents nominal excess propellants as a function of the probability of sufficient fuel aboard Centaur to perform the mission.

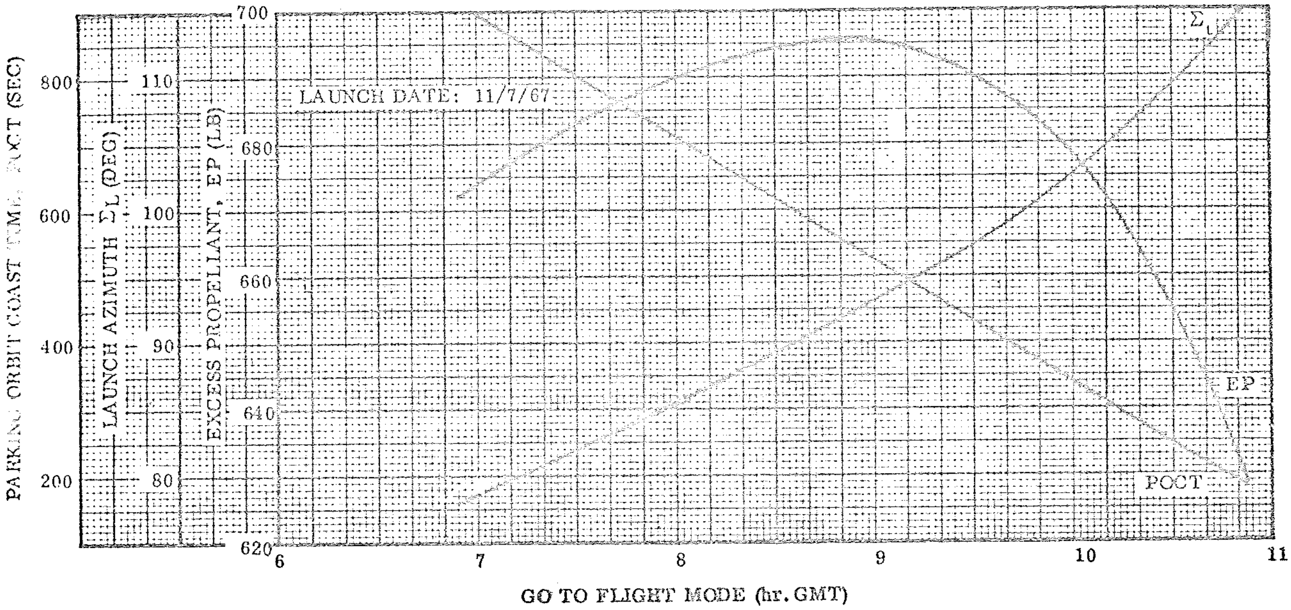


Figure 1-6. Typical Launch Window

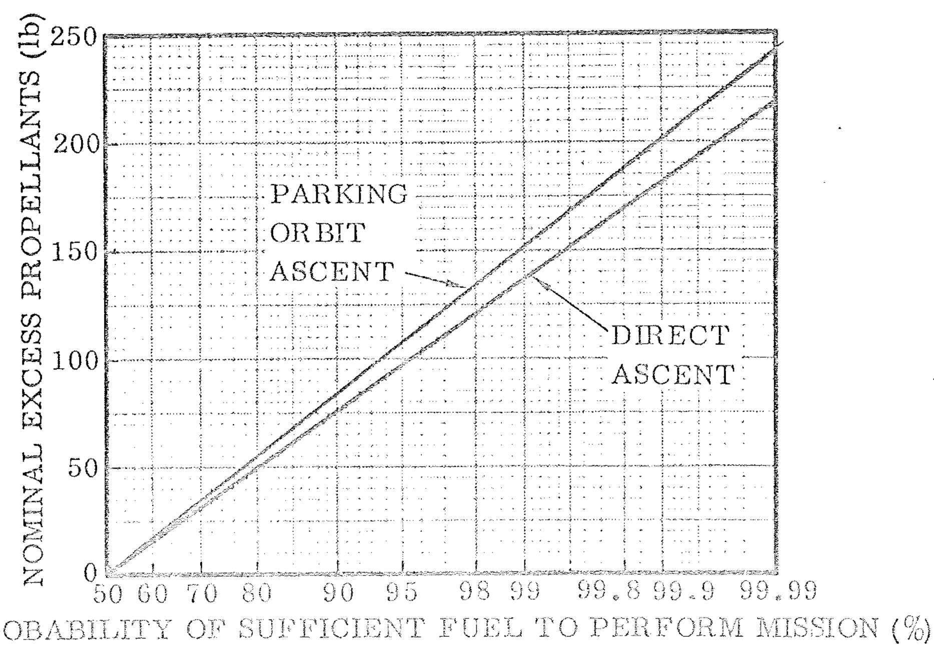


Figure 1-7. Nominal Excess Propellants Versus Percent Probability of Sufficient Fuel Remaining

Firing window data, based on targeted results are presented in Table 1-3.

Table 1-3. Firing Window Data AC-14

LAUNCH DATE	FIRING WINDOW OPENING CONDITIONS[1]			FIRING WINDOW CLOSING CONDITIONS[1]			WINDOW LENGTH (min)
	TIME[3] (GMT, hr: min)	AZIMUTH (deg)	POCT[2] (min)	TIME[3] (GMT, hr: min)	AZIMUTH (deg)	POCT[2] (min)	
5 November 1967	6:47	78.039	9.490	9:12	93.862	1.942	145
6 November 1967 [4]	6:53	78.054	12.265	10:13	106.165	1.957	197
7 November 1967	6:55	78.036	15.128	10:53	114.945	2.992	238
7 November 1967	6:55	78.014	15.171	10:53	114.854	3.033	238
8 November 1967	6:56	78.025	18.036	11:03	114.768	5.446	247
9 November 1967	6:56	78.028	20.881	11:12	114.880	7.840	256
10 November 1967 [5]	6:56	78.080	23.668	11:20	114.914	10.230	264
11 November 1967	7:24	82.255	24.970	11:28	114.918	12.575	244
12 November 1967	8:16	89.645	24.992	11:37	114.917	14.848	201

[1] See definition in Section 1.3.4.
[2] Parking orbit coast time (POCT) is defined as the time interval from first Centaur main engine cutoff to second Centaur main engine start.
[3] Liftoff time (two-inch motion).
[4] Site A: 0.83°N Latitude and 24.00°E Longitude.
[5] Site B: 0.42°N Latitude and 1.33°W Longitude.

1.3.5 TARGETED DATA

The information presented herein was obtained from closed-loop guidance targeted trajectory simulation data contained in Reference 2. The AC-14 target criteria data, based on References 3 and 3.1, are presented in Table 1-4.

Table 1-4. Target Criteria for AC-14

LAUNCH DATE			LANDING LOCATION SELENOGRAPHIC		IMPACT SPEED	ARRIVAL TIME CONSTRAINTS EARLIEST ARRIVAL				
			LATITUDE (deg)	LONGITUDE (deg)	(m/sec)	DATE			GMT	
DAY	MONTH	YEAR				DAY	MONTH	YEAR	HR	MIN
5	Nov.	1967	0.83N	24.0E	2635	7	Nov.	1967	22	53
6	Nov.	1967	0.83N	24.0E	2635	8	Nov.	1967	23	30
7	Nov.	1967	0.83N	24.0E	2635	9	Nov.	1967	23	58
7	Nov.	1967	0.42N	1.33W	2635	9	Nov.	1967	23	58
8	Nov.	1967	0.42N	1.33W	2635	11	Nov.	1967	00	24
9	Nov.	1967	0.42N	1.33W	2635	12	Nov.	1967	00	54
10	Nov.	1967	0.42N	1.33W	2635	13	Nov.	1967	01	24
11	Nov.	1967	0.42N	1.33W	2635	14	Nov.	1967	01	49
12	Nov.	1967	0.42N	1.33W	2635	15	Nov.	1967	02	15

1.4 TRAJECTORY PARAMETERS

Nominal ascent trajectory data for this flight are presented in Figures 1-8 and 1-9. These data reflect typical tracjetory characteristics and are based on the August 1967 issue of the Monthly Configuration, Performance, and Weights Status Report.[1]

The depicted trajectory parameters in Figures 1-8 and 1-9 and the associated definitions are presented in Table 1-5.

[1] Appendix A, AC-14 (two burn, closed loop) trajectory simulation.

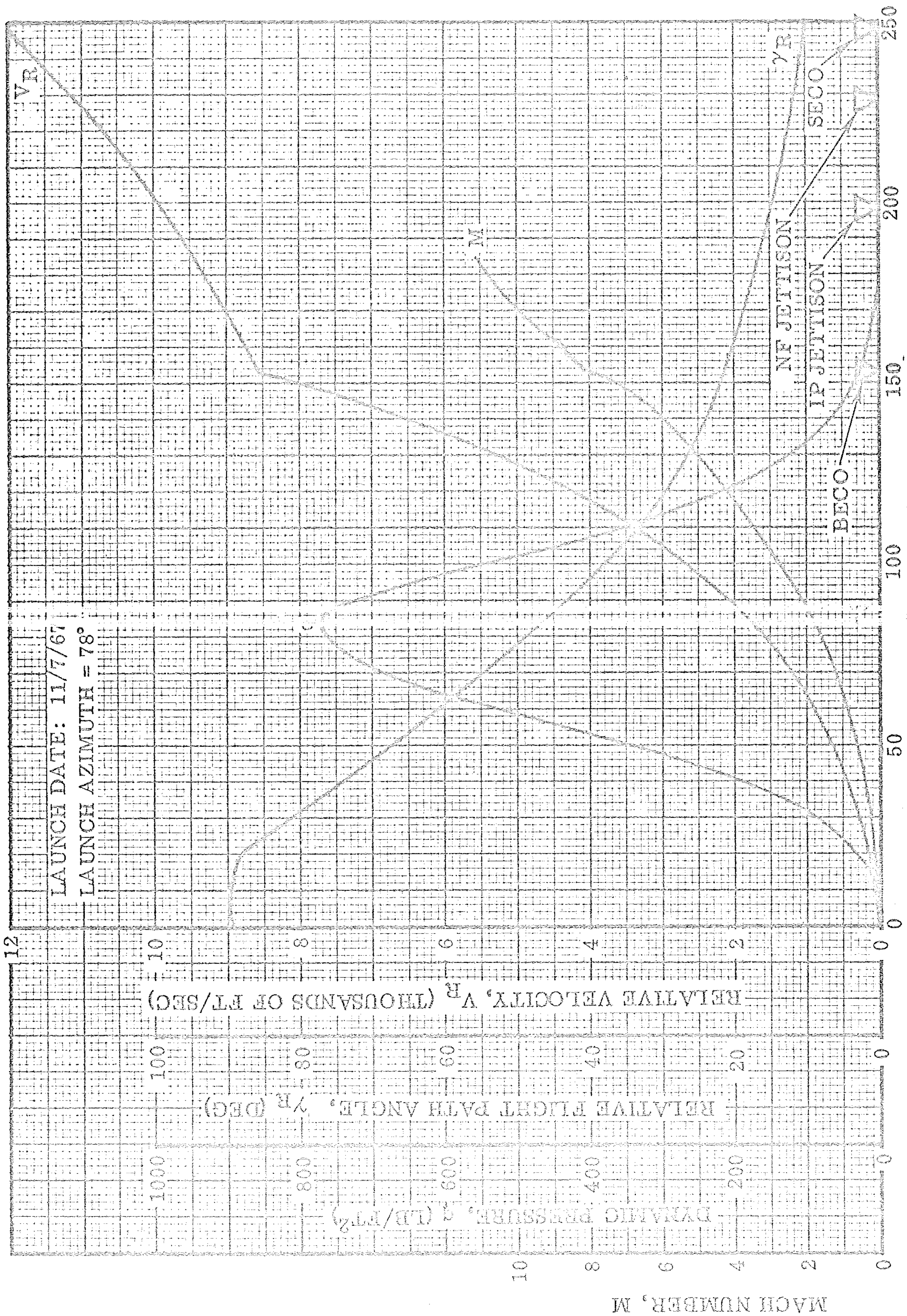


Figure 1-8. Atlas Phase Trajectory Parameters (Sheet 1 of 2)

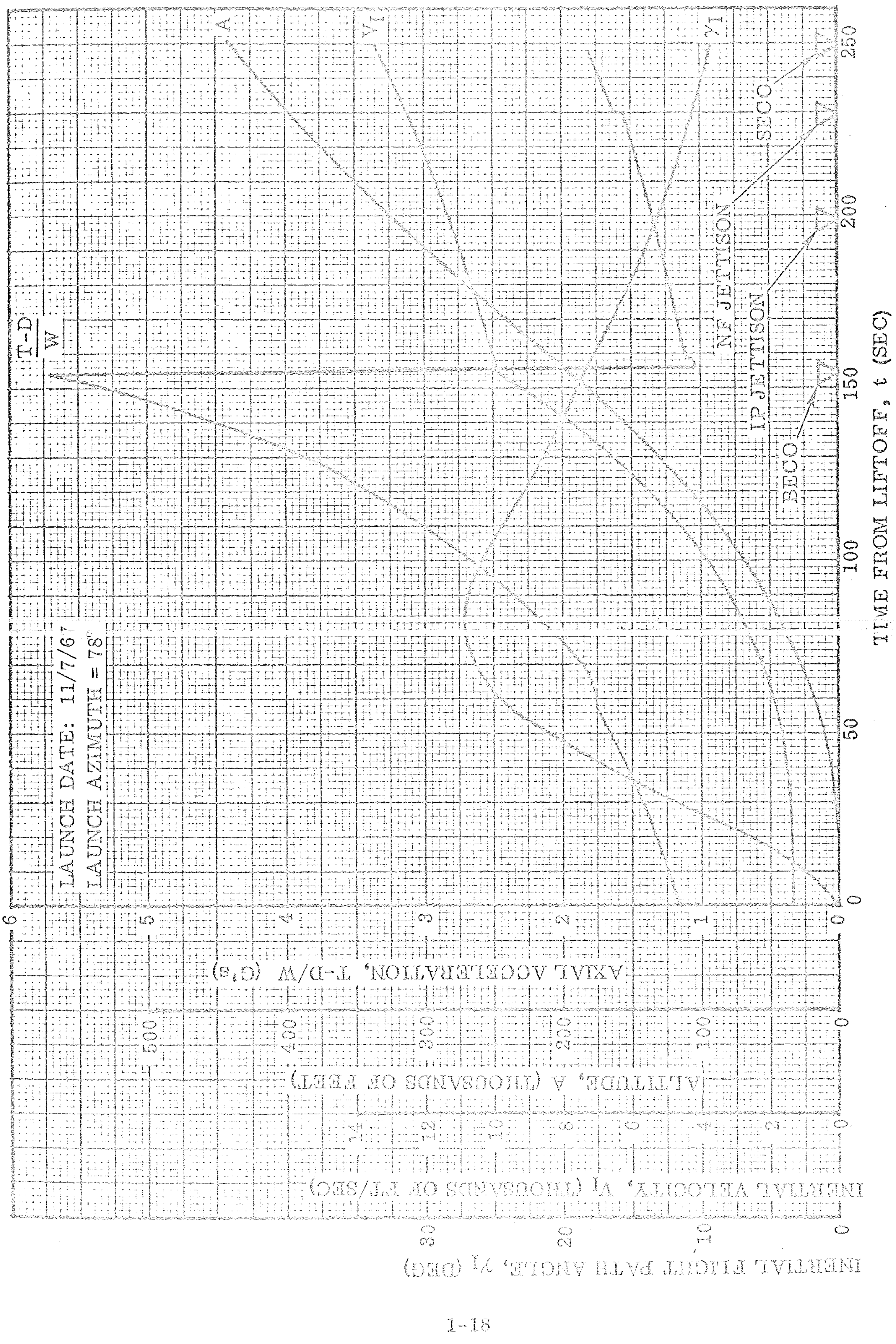


Figure 1-8. Atlas Phase Trajectory Parameters (Sheet 2 of 2)

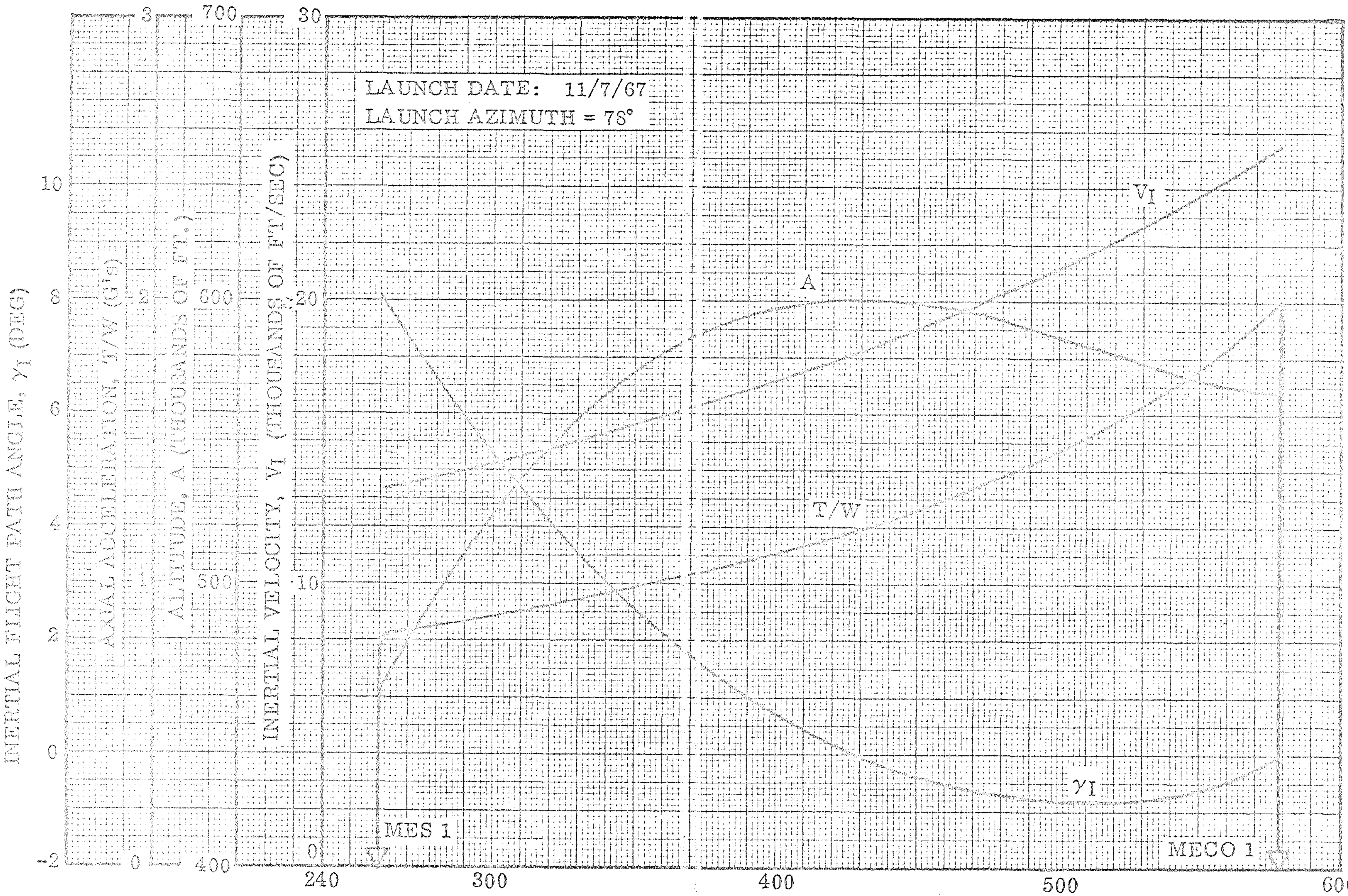


Figure 1-9. Centaur Phase Trajectory Parameters (Sheet 1 of 2)

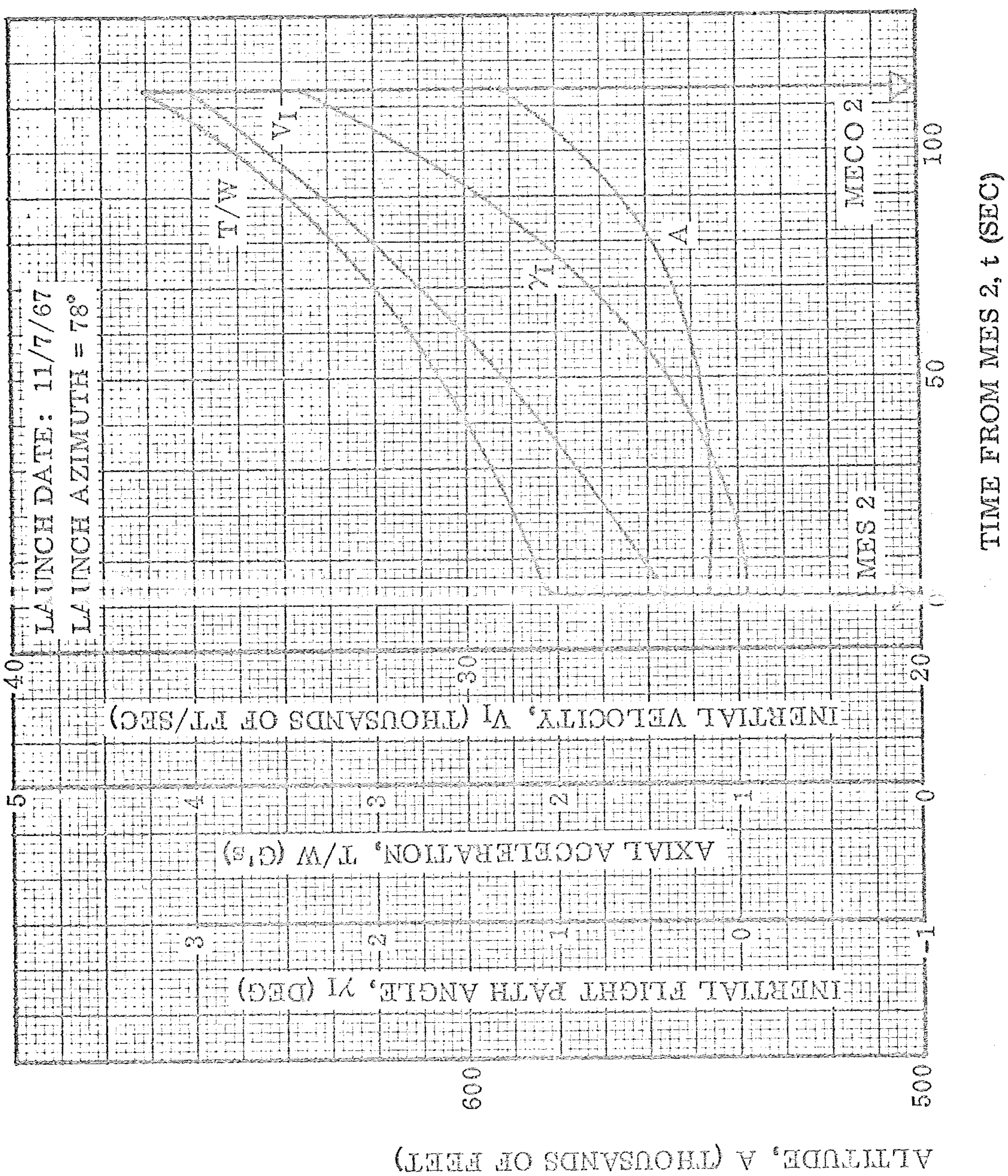


Figure 1-9. Centaur Phase Trajectory Parameters (Sheet 2 of 2)

Table 1-5. Trajectory Parameters

PARAMETER	DEFINITION
Altitude	Geocentric altitude above reference ellipsoid
Relative Velocity	Vehicle speed relative to the air
Inertial Velocity	Vehicle speed relative to a space fixed coordinate system
Flight Path Angle	Angle between the relative or inertial velocity vector and the local horizontal plane[1], measured positive above and negative below the horizontal plane
Axial Acceleration	The longitudinal component of thrust less drag divided by weight
Dynamic Pressure	Aerodynamic loading term equal to one-half the product of ambient air density and airstream velocity squared
Mach Number	Ratio of vehicle speed (with respect to air) to the speed of sound

[1] The local horizontal plane is a plane normal to the geocentric position vector.

1.5 ORBITAL ELEMENTS

Nominal orbital elements of a) parking orbit after initial ullage settling phase, b) spacecraft at time of separation from Centaur, and c) expended Centaur stage at end of retromaneuver are summarized in Tables 1-6, 1-7, and 1-8 respectively. Figure 1-10 presents orbital parameter definitions.

1.6 CENTAUR RETROMANEUVER

For operational Surveyor, the spacecraft will be oriented with respect to the sun and the star Canopus. Sun orientation will be accomplished shortly after spacecraft separation, following acquisition by the Johannesburg Deep Space Instrumentation Facility (DSIF). Star orientation will not be initiated until the separation distance between the spacecraft and Centaur is sufficient to preclude erroneous acquisition of the Centaur stage. Current separation requirements specify a minimum separation distance of 336 kilometers no later than 5 hours after injection. In order to meet this requirement, Centaur will be rotated approximately 180 degrees to the injection velocity vector and a

Table 1-6. Parking Orbit Nominal Orbital Element After Initial Ullage Settling Phase

COLUMN 1- LAUNCH DATE (M/D/Y)
COLUMN 2- FLIGHT AZIMUTH (DEG)
COLUMN 3- EPOCH (SECONDS FROM LIFTOFF)
COLUMN 4- SEMIMAJOR AXIS (N.MI.)
COLUMN 5- ECCENTRICITY
COLUMN 6- INCLINATION (DEG)
COLUMN 7- LONGITUDE OF ASCENDING NODE (DEG)
COLUMN 8- ARGUMENT OF PERIGEE (DEG)
COLUMN 9- TRUE ANOMALY (DEG)
COLUMN 10- PERIGEE ALTITUDE (N.MI.)
COLUMN 11- APOGEE ALTITUDE (N.MI.)
COLUMN 12- PERIOD (DAYS)
COLUMN 13- COAST TIME, MECO1 TO MES2 (MIN)

	1	2	3	4	5	6	7	8	9	10	11	12	13
	11/ 5/67	78	655.62	3531.00	0.00104	30.29	290.76	284.79	-189.95	83.38	90.74	0.061	9.52
	11/ 5/67	81	655.62	3531.37	0.00095	29.43	285.33	291.34	-191.76	84.07	90.78	0.061	7.83
	11/ 5/67	84	655.62	3531.73	0.00086	28.81	279.64	298.49	-193.94	84.75	90.82	0.061	6.22
	11/ 5/67	87	655.62	3532.08	0.00077	28.43	273.77	306.31	-196.60	85.41	90.87	0.061	4.70
	11/ 5/67	90	655.62	3532.42	0.00069	28.31	267.81	314.84	-199.88	86.04	90.91	0.061	3.37
	11/ 5/67	93	654.77	3532.69	0.00063	28.45	261.69	323.67	-203.52	86.54	90.98	0.061	2.26
	11/ 6/67	78	655.62	3531.00	0.00104	30.29	290.76	284.79	-189.95	83.38	90.74	0.061	12.30
A	11/ 6/67	81	655.62	3531.37	0.00095	29.43	285.33	291.34	-191.76	84.07	90.78	0.061	10.80
	11/ 6/67	84	655.62	3531.73	0.00086	28.81	279.64	298.49	-193.94	84.75	90.82	0.061	9.33
	11/ 6/67	87	655.62	3532.08	0.00077	28.43	273.77	306.31	-196.60	85.41	90.87	0.061	7.95
	11/ 6/67	90	655.62	3532.42	0.00069	28.31	267.81	314.84	-199.88	86.04	90.91	0.061	6.66
	11/ 6/67	93	655.62	3532.74	0.00061	28.45	261.85	324.20	-203.99	86.65	90.96	0.061	5.50
	11/ 6/67	96	655.62	3533.06	0.00054	28.84	255.99	334.51	-209.17	87.22	91.02	0.061	4.47
	11/ 6/67	99	653.62	3533.36	0.00047	29.49	250.32	346.07	-215.77	87.76	91.09	0.061	3.60
	11/ 6/67	102	655.62	3533.65	0.00042	30.36	244.90	359.02	-224.11	88.24	91.18	0.061	2.86
	11/ 6/67	105	655.51	3533.92	0.00037	31.44	239.78	13.66	125.70	88.67	91.30	0.061	2.21
	11/ 6/67	106	655.69	3534.02	0.00036	31.85	238.15	19.17	121.57	88.81	91.35	0.061	2.00
	11/ 7/67	78	655.62	3531.00	0.00104	30.29	290.76	284.79	-189.95	83.38	90.74	0.061	15.18
	11/ 7/67	81	655.62	3531.37	0.00095	29.43	285.33	291.34	-191.76	84.07	90.78	0.061	13.80
	11/ 7/67	84	655.62	3531.73	0.00086	28.81	279.64	298.49	-193.94	84.75	90.82	0.061	12.47
	11/ 7/67	87	655.62	3532.08	0.00077	28.43	273.77	306.31	-196.60	85.41	90.87	0.061	11.18
	11/ 7/67	90	655.62	3532.42	0.00069	28.31	267.81	314.84	-199.88	86.04	90.91	0.061	9.90
	11/ 7/67	93	655.62	3532.74	0.00061	28.45	261.85	324.20	-203.99	86.65	90.96	0.061	8.71
	11/ 7/67	96	655.62	3533.06	0.00054	28.84	255.99	334.51	-209.17	87.22	91.02	0.061	7.81
	11/ 7/67	99	653.62	3533.36	0.00047	29.49	250.32	346.07	-215.77	87.76	91.09	0.061	6.62
	11/ 7/67	102	655.62	3533.65	0.00042	30.36	244.90	359.02	-224.11	88.24	91.18	0.061	5.74
	11/ 7/67	105	655.62	3533.93	0.00037	31.44	239.78	13.75	125.60	88.67	91.30	0.061	4.98
	11/ 7/67	108	656.76	3534.27	0.00034	32.72	234.98	33.84	109.62	89.12	91.54	0.061	4.30
	11/ 7/67	111	656.76	3534.52	0.00034	34.17	230.53	50.59	96.57	89.38	91.77	0.061	3.71
	11/ 7/67	114	657.90	3534.83	0.00035	35.77	226.40	69.64	80.94	89.64	92.13	0.061	3.16
	11/ 7/67	115	657.90	3534.90	0.00036	36.33	225.09	74.39	77.24	89.69	92.24	0.061	3.03
	11/ 8/67	78	655.62	3531.00	0.00104	30.29	290.76	284.79	-189.95	83.38	90.74	0.061	18.06
	11/ 8/67	81	655.62	3531.37	0.00095	29.43	285.33	291.34	-191.76	84.07	90.78	0.061	16.80
	11/ 8/67	84	655.62	3531.73	0.00086	28.81	279.64	298.49	-193.94	84.75	90.82	0.061	15.53
	11/ 8/67	87	655.62	3532.08	0.00077	28.43	273.77	306.31	-196.60	85.41	90.87	0.061	14.26
	11/ 8/67	90	655.62	3532.42	0.00069	28.31	267.81	314.84	-199.88	86.04	90.91	0.061	13.00
	11/ 8/67	93	655.62	3532.74	0.00061	28.45	261.85	324.20	-203.99	86.65	90.96	0.061	11.80
	11/ 8/67	96	655.62	3533.06	0.00054	28.84	255.99	334.51	-209.17	87.22	91.02	0.061	10.65
	11/ 8/67	99	655.62	3533.36	0.00047	29.49	250.32	346.07	-215.77	87.76	91.09	0.061	9.60
	11/ 8/67	102	655.62	3533.65	0.00042	30.36	244.90	359.02	-224.11	88.24	91.18	0.061	8.63
	11/ 8/67	105	655.62	3533.93	0.00037	31.44	239.78	13.75	125.60	88.67	91.30	0.061	7.76
	11/ 8/67	108	656.76	3534.27	0.00034	32.72	234.98	33.84	109.62	89.12	91.54	0.061	6.96
	11/ 8/67	111	656.76	3534.52	0.00034	34.17	230.53	50.59	96.57	89.38	91.77	0.061	6.26
	11/ 8/67	114	657.90	3534.83	0.00035	35.77	226.40	69.64	80.94	89.64	92.13	0.061	5.61
	11/ 8/67	115	657.90	3534.90	0.00036	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
	11/ 9/67	78	655.62	3531.00	0.00104	30.29	290.76	284.79	-189.95	83.38	90.74	0.061	20.91
	11/ 9/67	81	655.62	3531.37	0.00095	29.43	285.33	291.34	-191.76	84.07	90.78	0.061	19.73
	11/ 9/67	84	655.62	3531.73	0.00086	28.81	279.64	298.49	-193.94	84.75	90.82	0.061	18.51
	11/ 9/67	87	655.62	3532.08	0.00077	28.43	273.77	306.31	-196.60	85.41	90.87	0.061	17.28
	11/ 9/67	90	655.62	3532.42	0.00069	28.31	267.81	314.84	-199.88	86.04	90.91	0.061	16.04
	11/ 9/67	93	655.62	3532.74	0.00061	28.45	261.85	324.20	-203.99	86.65	90.96	0.061	14.83
	11/ 9/67	96	655.62	3533.06	0.00054	28.84	255.99	334.51	-209.17	87.22	91.02	0.061	13.63
	11/ 9/67	99	655.62	3533.36	0.00047	29.49	250.32	346.07	-215.77	87.76	91.09	0.061	12.51
	11/ 9/67	102	655.62	3533.55	0.00042	30.36	244.90	359.02	-224.11	88.24	91.18	0.061	11.46
	11/ 9/67	105	655.62	3533.93	0.00037	31.44	239.78	13.75	125.60	88.67	91.30	0.061	10.49
	11/ 9/67	108	656.76	3534.27	0.00034	32.72	234.98	33.84	109.62	89.12	91.54	0.061	9.60
	11/ 9/67	111	656.76	3534.52	0.00034	34.17	230.53	50.59	96.57	89.38	91.77	0.061	8.80
B	11/ 9/67	114	657.90	3534.83	0.00035	35.77	226.40	69.64	80.94	89.64	92.13	0.061	8.04
	11/ 9/67	115	657.90	3534.90	0.00036	36.33	225.09	74.39	77.24	89.69	92.24	0.061	7.81
	11/10/67	78	655.62	3531.00	0.00104	30.29	290.76	284.79	-189.95	83.38	90.74	0.061	23.72
	11/10/67	81	655.62	3531.37	0.00095	29.43	285.33	291.34	-191.76	84.07	90.78	0.061	22.60
	11/10/67	84	655.62	3531.73	0.00086	28.81	279.64	298.49	-193.94	84.75	90.82	0.061	21.44
	11/10/67	87	655.62	3532.08	0.00077	28.43	273.77	306.31	-196.60	85.41	90.87	0.061	20.24
	11/10/67	90	655.62	3532.42	0.00069	28.31	267.81	314.84	-199.88	86.04	90.91	0.061	19.02
	11/10/67	93	655.62	3532.74	0.00061	28.45	261.85	324.20	-203.99	86.65	90.96	0.061	17.79
	11/10/67	96	655.62	3533.06	0.00054	28.84	255.99	334.51	-209.17	87.22	91.02	0.061	16.55
	11/10/67	99	655.62	3533.36	0.00047	29.49	250.32	346.07	-215.77	87.76	91.09	0.061	15.38
	11/10/67	102	655.62	3533.65	0.00042	30.36	244.90	359.02	-224.11	88.24	91.18	0.061	14.26
	11/10/67	105	655.62	3533.93	0.00037	31.44	239.78	13.75	125.60	88.67	91.30	0.061	13.21
	11/10/67	108	656.76	3534.27	0.00034	32.72	234.98	33.84	109.62	89.12	91.54	0.061	12.22
	11/10/67	111	656.76	3534.52	0.00034	34.17	230.53	50.59	96.57	89.38	91.77	0.061	11.31
	11/10/67	114	657.90	3534.33	0.00035	35.77	226.40	69.64	80.94	89.64	92.13	0.061	10.47
	11/10/67	115	657.90	3534.90	0.00036	36.33	225.09	74.39	77.24	89.69	92.24	0.061	10.21
	11/11/67	82	655.62	3531.49	0.00092	29.20	283.46	293.65	-192.44	84.30	90.80	0.061	25.09
	11/11/67	84	655.62	3531.73	0.00086	28.81	279.64	298.49	-193.94	84.75	90.82	0.061	24.34
	11/11/67	87	655.62	3532.08	0.00077	28.43	273.77	306.31	-196.60	85.41	90.87	0.061	23.17
	11/11/67	90	655.62	3532.42	0.00069	28.31	267.81	314.84	-199.88	86.04	90.91	0.061	21.95
	11/11/67	93	655.62	3532.74	0.00061	28.45	261.85	324.20	-203.99	86.65	90.96	0.061	20.72
	11/11/67	96	655.62	3533.06	0.00054	28.84	255.99	334.51	-209.17	87.22	91.02	0.061	19.44
	11/11/67	99	655.62	3533.36	0.00047	29.49	250.32	346.07	-215.77	87.76	91.09	0.061	18.21
	11/11/67	102	655.62	3533.65	0.00042	30.36	244.90	359.02	-224.11	88.24	91.18	0.061	17.01
	11/11/67	105	655.62	3533.93	0.00037	31.44	239.78	13.75	125.60	88.67	91.30	0.061	15.87
	11/11/67	108	656.76	3534.27	0.00034	32.72	234.98	33.84	109.62	89.12	91.54	0.061	14.79
	11/11/67	111	656.76	3534.52	0.00034	34.17	230.53	50.59	96.57	89.38	91.77	0.061	13.80
	11/11/67	114	657.90	3534.83	0.00035	35.77	226.40	69.64	80.94	89.64	92.13	0.061	12.85
	11/11/67	115	657.90	3534.90	0.00036	36.33	225.09	74.39	77.24	89.69	92.24	0.061	12.56
	11/12/67	89	655.62	3532.30	0.00072	28.32	269.80	311.91	-198.70	85.83	90.90	0.061	25.28
	11/12/67	90	655.62	3532.42	0.00069	28.31	267.81	314.84	-199.88	86.04	90.91	0.061	24.86
	11/12/67	93	655.62	3532.74	0.00061	28.45	261.85	324.20	-203.99	86.65	90.96	0.061	23.60
	11/12/67	96	655.62	3533.06	0.00054	28.84	255.99	334.51	-209.17	87.22	91.02	0.061	22.31
	11/12/67	99	655.62	3533.36	0.00047	29.49	250.32	346.07	-215.77	87.76	91.09	0.061	21.02
	11/12/67	102	655.62	3533.65	0.00042	30.36	244.90	359.02	-224.11	88.24	91.18	0.061	19.73
	11/12/67	105	655.62	3533.93	0.00037	31.44	239.78	13.75	125.60	88.67	91.30	0.061	18.49
	11/12/67	108	656.76	3534.27	0.00034	32.72	234.98	33.84	109.62	89.12	91.54	0.061	17.30
	11/12/67	111	656.76	3534.52	0.00034	34.17	230.53	50.59	96.57	89.38	91.77	0.061	16.19
	11/12/67	114	657.90	3534.83	0.00035	35.77	226.40	69.64	80.94	89.64	92.13	0.061	15.15
	11/12/67	115	657.90	3534.90	0.00036	36.33	225.09	74.39	77.24	89.69	92.24	0.061	14.83

SITE A, 0.83°N LATITUDE AND 24.0°E LONGITUDE. SITE B, 0.42°N LATITUDE AND 1.33°W LONGITUDE.

Table 1-7. Spacecraft Nominal Orbital Elements at Time of Separation from Centaur

COLUMN 1- LAUNCH DATE (M/D/Y)
COLUMN 2- FLIGHT AZIMUTH (DEG)
COLUMN 3- EPOCH (SECONDS FROM LIFTOFF)
COLUMN 4- SEMIMAJOR AXIS (N.MI.)
COLUMN 5- ECCENTRICITY
COLUMN 6- INCLINATION (DEG)
COLUMN 7- LONGITUDE OF ASCENDING NODE (DEG)
COLUMN 8- ARGUMENT OF PERIGEE (DEG)
COLUMN 9- TRUE ANOMALY (DEG)
COLUMN 10- PERIGEE ALTITUDE (N.MI.)
COLUMN 11- APOGEE ALTITUDE (N.MI.)
COLUMN 12- PERIOD (DAYS)

	1	2	3	4	5	6	7	8	9	10	11	12
	11/ 5/67	78	1327.10	164366.94	0.97849	30.30	287.69	133.02	10.79	91.92	~521754.08	19.345
	11/ 5/67	81	1225.64	167769.76	0.97892	29.44	282.90	130.85	10.73	92.21	328559.42	19.949
	11/ 5/67	84	1128.74	171365.65	0.97936	28.84	277.57	129.18	10.80	92.42	335750.99	20.594
	11/ 5/67	87	1037.56	173881.07	0.97966	28.44	272.15	128.06	10.80	92.57	340781.69	21.049
	11/ 5/67	90	957.74	174097.55	0.97969	28.32	266.51	127.88	10.78	92.64	341194.59	21.087
	11/ 5/67	93	890.48	174243.31	0.97970	28.43	260.91	128.44	10.80	92.65	341506.09	21.115
	11/ 6/67	78	1493.54	175511.24	0.97986	30.34	287.13	144.45	10.79	91.46	344043.13	21.346
	11/ 6/67	81	1403.48	179054.55	0.98025	29.39	282.22	142.89	10.79	91.80	351129.42	21.996
A	11/ 6/67	84	1315.70	181962.25	0.98057	28.87	276.74	142.05	10.76	92.09	356944.53	22.534
	11/ 6/67	87	1232.48	182039.45	0.98057	28.50	271.22	141.52	10.75	92.32	357098.69	22.548
	11/ 6/67	90	1154.96	182259.95	0.98060	28.30	265.73	141.33	10.77	92.48	357539.52	22.589
	11/ 6/67	93	1085.42	182252.04	0.98060	28.40	260.13	141.73	10.80	92.60	357523.59	22.587
	11/ 6/67	96	1023.86	182374.82	0.98061	28.83	254.47	142.75	10.77	92.67	357769.08	22.610
	11/ 6/67	99	971.42	182527.15	0.98062	29.53	248.93	144.19	10.77	92.70	358073.72	22.639
	11/ 6/67	102	926.96	182627.19	0.98063	30.40	243.70	145.79	10.81	92.69	358273.80	22.657
	11/ 6/67	105	888.20	182679.18	0.98064	31.35	238.92	147.37	10.82	92.67	358377.80	22.667
	11/ 6/67	106	875.66	182807.09	0.98065	31.68	237.43	147.87	10.76	92.67	358633.63	22.691
	11/ 7/67	78	1666.82	187833.52	0.98118	30.39	286.37	156.31	10.81	90.88	368668.26	23.633
	11/ 7/67	81	1583.60	188445.22	0.98124	29.28	281.54	155.15	10.79	91.25	369911.30	23.748
	11/ 7/67	84	1503.80	188594.67	0.98125	28.82	276.04	154.81	10.78	91.58	370209.88	23.777
	11/ 7/67	87	1426.28	188825.86	0.98127	28.57	270.37	154.74	10.82	91.87	370671.96	23.820
	11/ 7/67	90	1349.90	188942.44	0.98128	28.43	264.76	154.79	10.77	92.13	370904.87	23.842
	11/ 7/67	93	1278.08	188995.56	0.98129	28.46	259.22	155.02	10.77	92.33	371010.90	23.853
	11/ 7/67	96	1211.96	189006.15	0.98129	28.76	253.73	155.56	10.77	92.48	371031.92	23.855
	11/ 7/67	99	1152.68	189022.98	0.98129	29.39	248.32	156.44	10.77	92.59	371065.49	23.858
	11/ 7/67	102	1100.24	189051.79	0.98129	30.32	243.06	157.60	10.76	92.66	371123.03	23.863
	11/ 7/67	105	1054.64	189081.23	0.98130	31.51	238.06	158.91	10.78	92.69	371181.91	23.869
	11/ 7/67	108	1014.74	189036.52	0.98129	32.85	233.39	160.24	10.80	92.71	371092.43	23.860
	11/ 7/67	111	979.40	188948.95	0.98128	34.30	229.10	161.53	10.79	92.71	370917.31	23.844
	11/ 7/67	114	948.62	188826.27	0.98127	35.79	225.17	162.72	10.79	92.69	370671.96	23.820
	11/ 7/67	115	939.50	188769.88	0.98126	36.30	223.94	163.09	10.81	92.67	370559.19	23.810
	11/ 8/67	78	1838.96	191320.03	0.98153	30.33	285.71	168.00	10.84	90.24	375661.93	24.294
	11/ 8/67	81	1763.72	191290.05	0.98152	29.34	280.66	167.50	10.86	90.61	375601.61	24.288
	11/ 8/67	84	1687.34	191348.92	0.98153	28.79	275.27	167.30	10.83	90.97	375718.99	24.299
	11/ 8/67	87	1610.96	191524.38	0.98154	28.50	269.69	167.30	10.81	91.31	376069.57	24.333
	11/ 8/67	90	1535.72	191460.35	0.98153	28.41	264.04	167.43	10.79	91.62	375941.19	24.321
	11/ 8/67	93	1463.90	191447.60	0.98153	28.52	258.41	167.69	10.83	91.89	375915.41	24.318
	11/ 8/67	96	1394.36	191430.56	0.98153	28.84	252.88	168.10	10.77	92.13	375881.11	24.315
	11/ 8/67	99	1331.66	191524.09	0.98154	29.45	247.49	168.68	10.83	92.31	376067.99	24.333
	11/ 8/67	102	1273.52	191433.65	0.98153	30.33	242.31	169.41	10.82	92.45	375886.96	24.316
	11/ 8/67	105	1221.08	191423.05	0.98153	31.46	237.40	170.22	10.82	92.55	375865.67	24.314
	11/ 8/67	108	1174.34	191349.52	0.98152	32.79	232.79	171.06	10.83	92.64	375718.51	24.300
	11/ 8/67	111	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	10.83	92.68	375429.46	24.272
	11/ 8/67	114	[illegible]	[illegible]	0.98148	35.81	226.56	172.65	10.83	92.70	374962.66	[illegible]
	11/ 8/67	115	1083.14	190871.23	0.98147	36.34	[illegible]	[illegible]	10.84	[illegible]	[illegible]	[illegible]
	11/ 9/67	78	2009.96	191154.88	0.98151	30.33	265.00	179.66	10.89	89.58	375332.29	24.262
	11/ 9/67	81	1939.28	191154.93	0.98151	29.40	279.87	179.55	10.88	89.94	375332.03	24.262
	11/ 9/67	84	1866.32	191176.11	0.98151	28.81	274.50	179.56	10.85	90.31	375374.03	24.267
	11/ 9/67	87	1792.22	191183.46	0.98151	28.48	268.95	179.65	10.84	90.67	375388.37	24.268
	11/ 9/67	90	1718.12	191176.78	0.98151	28.39	263.31	179.82	10.84	91.02	375374.65	24.267
	11/ 9/67	93	1645.16	191130.16	0.98150	28.52	257.67	180.04	10.86	91.35	375281.09	24.258
	11/ 9/67	96	1573.34	191095.90	0.98150	28.88	252.12	180.32	10.80	91.65	375212.27	24.251
	11/ 9/67	99	1506.08	191011.79	0.98149	29.47	246.73	180.66	10.80	91.90	375043.80	24.235
	11/ 9/67	102	1443.38	191071.32	0.98149	30.34	241.58	181.05	10.81	92.11	375162.64	24.247
	11/ 9/67	105	1385.24	190913.24	0.98148	31.46	236.71	181.46	10.80	92.28	374846.32	24.216
	11/ 9/67	108	1332.60	190784.95	0.98146	32.78	232.19	181.87	10.83	92.44	374589.58	24.192
	11/ 9/67	111	1284.92	190589.76	0.98144	34.25	227.90	182.27	10.85	92.53	374199.09	24.155
B	11/ 9/67	114	1240.46	190280.70	0.98141	35.81	223.96	182.65	10.79	92.62	373580.89	24.096
	11/ 9/67	115	1226.78	190149.47	0.98140	36.35	222.71	182.77	10.78	92.63	373318.43	24.071
	11/10/67	78	2178.68	188758.67	0.98128	30.33	284.29	191.15	10.95	88.96	370540.49	23.808
	11/10/67	81	2111.42	188789.16	0.98128	29.43	279.13	191.36	10.87	89.30	370601.13	23.813
	11/10/67	84	2041.68	188772.17	0.98128	28.82	273.75	191.58	10.86	89.66	370566.79	23.810
	11/10/67	87	1970.06	188759.18	0.98128	28.48	268.21	191.79	10.86	90.02	370540.45	23.808
	11/10/67	90	1897.10	188708.87	0.98127	28.38	262.58	191.99	10.90	90.39	370439.45	23.798
	11/10/67	93	1823.00	188642.46	0.98126	28.52	256.94	192.16	10.89	90.75	370306.29	23.786
	11/10/67	96	1748.90	188585.38	0.98125	28.87	251.38	192.31	10.82	91.10	370191.76	23.775
	11/10/67	99	1678.22	188490.36	0.98124	29.49	246.01	192.42	10.81	91.41	370001.43	23.757
	11/10/67	102	1610.96	188473.51	0.98124	30.36	240.87	192.51	10.81	91.68	369967.46	23.754
	11/10/67	105	1548.26	188222.56	0.98121	31.46	236.03	192.56	10.84	91.90	369465.34	23.706
	11/10/67	108	1490.12	188027.55	0.98119	32.78	231.51	192.56	10.87	92.14	369075.02	23.670
	11/10/67	111	1435.40	187769.95	0.98117	34.24	227.29	192.58	10.81	92.30	368559.71	23.621
	11/10/67	114	1386.38	187432.70	0.98113	35.82	223.36	192.57	10.83	92.43	367885.07	23.557
	11/10/67	115	1370.42	187312.06	0.98112	36.36	222.12	192.57	10.80	92.47	367643.76	23.535
	11/11/67	82	2260.76	185410.33	0.98095	29.19	276.62	203.14	10.97	88.83	363843.94	23.177
	11/11/67	84	2216.30	184988.53	0.98090	28.82	273.01	203.42	10.97	89.05	363000.13	23.098
	11/11/67	87	2145.62	184986.73	0.98090	28.48	267.48	203.76	10.90	89.41	362992.17	23.097
	11/11/67	90	2077.66	185628.42	0.98096	28.39	261.87	203.99	10.89	89.78	364273.16	23.218
	11/11/67	93	1998.56	184828.54	0.98088	28.52	256.23	204.11	10.93	90.15	362679.05	23.068
	11/11/67	96	1922.18	184769.53	0.98087	28.87	250.66	204.13	10.85	90.53	362560.64	23.057
	11/11/67	99	1848.08	184614.66	0.98085	29.49	245.27	204.01	10.85	90.88	362250.56	23.028
	11/11/67	102	1776.26	184463.81	0.98084	30.36	240.18	203.79	10.84	91.20	361948.54	23.000
	11/11/67	105	1707.66	184275.19	0.98081	31.46	235.37	203.48	10.83	91.48	361571.01	22.965
	11/11/67	108	1644.02	184034.47	0.98079	32.78	230.88	203.10	10.85	91.79	361089.27	22.920
	11/11/67	111	1584.74	183709.56	0.98075	34.23	226.69	202.69	10.88	91.99	360439.27	22.859
	11/11/67	114	1526.68	183314.67	0.98071	35.82	222.78	202.27	10.84	92.20	359649.65	22.785
	11/11/67	115	1511.78	183164.97	0.98069	36.36	221.53	202.16	10.86	92.24	359349.62	22.757
	11/12/67	89	2272.16	180559.23	0.98043	28.34	282.97	219.66	10.99	89.12	354141.45	22.273
	11/12/67	92	2257.08	180957.10	0.98044	28.35	281.10	219.93	10.93	89.25	354137.06	22.273
	11/12/67	93	2171.86	180434.12	0.98042	28.80	255.50	215.97	10.94	89.61	353500.74	22.251
	11/12/67	96	2094.93	180521.50	0.98040	28.90	250.79	215.19	10.93	89.99	353674.89	22.231
	11/12/67	99	2019.46	180199.05	0.98039	29.31	[illegible]	215.48	10.85	90.26	352921.41	22.207
	11/12/67	102	1939.28	180000.95	0.98036	30.37	[illegible]	214.87	10.90	90.72	[illegible]	22.170
	11/12/67	105	1865.18	179751.67	0.98034	31.45	236.72	214.18	10.89	91.04	352524.41	22.124
	11/12/67	108	1795.50	179450.18	0.98030	32.73	230.74	213.59	10.86	91.42	351931.05	22.068
	11/12/67	111	1729.38	179079.31	[illegible]	[illegible]	[illegible]	212.55	[illegible]	91.66	[illegible]	22.000
	11/12/67	114	1668.82	178744.72	0.98022	35.81	222.41	211.71	10.84	91.95	350509.03	21.938
	11/12/67	115	1647.44	178490.85	0.98019	36.36	220.97	211.44	10.84	91.99	350001.78	21.892

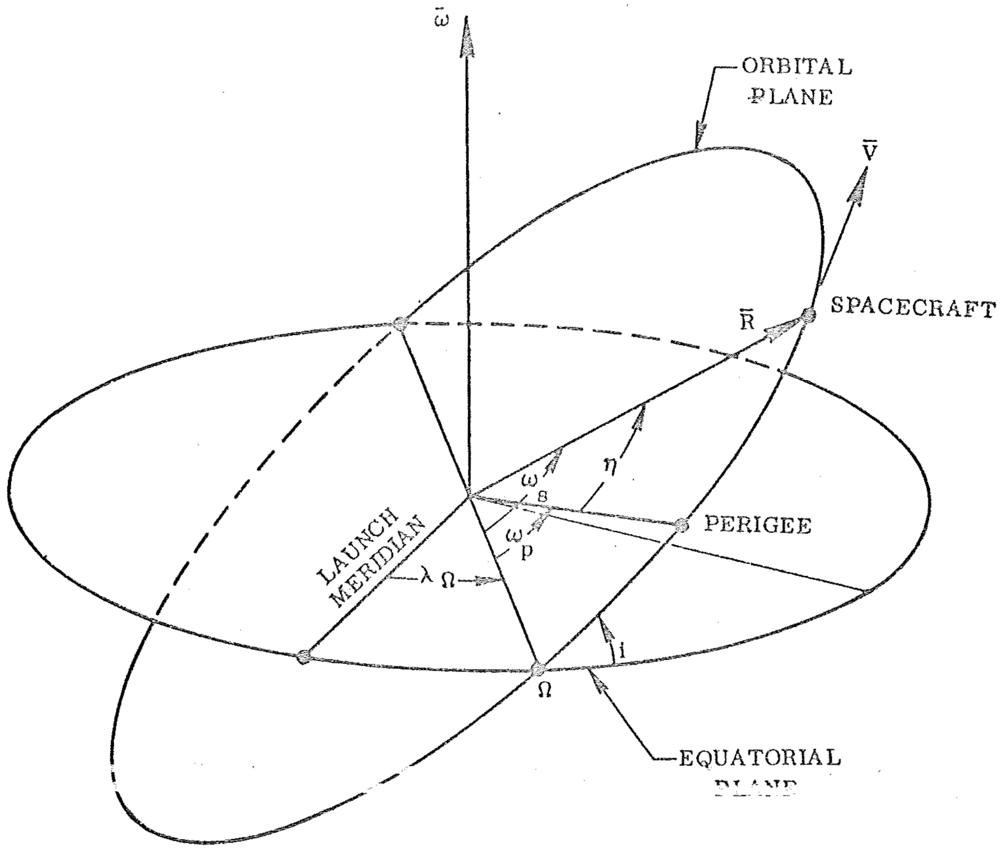
SITE A, 0.83°N LATITUDE AND 24.0°E LONGITUDE. SITE B, 0.42°N LATITUDE AND 1.33°W LONGITUDE.

Table 1-8. Expended Centaur Nominal Orbital Elements at End of Retromaneuver

COLUMN 1- LAUNCH DATE (M/D/Y)
COLUMN 2- FLIGHT AZIMUTH (DEG)
COLUMN 3- EPOCH (SECONDS FROM LIFTOFF)
COLUMN 4- SEMIMAJOR AXIS (N.MI.)
COLUMN 5- ECCENTRICITY
COLUMN 6- INCLINATION (DEG)
COLUMN 7- LONGITUDE OF ASCENDING NODE (DEG)
COLUMN 8- ARGUMENT OF PERIGEE (DEG)
COLUMN 9- TRUE ANOMALY (DEG)
COLUMN 10- PERIGEE ALTITUDE (N.MI.)
COLUMN 11- APOGEE ALTITUDE (N.MI.)
COLUMN 12- PERIOD (DAYS)

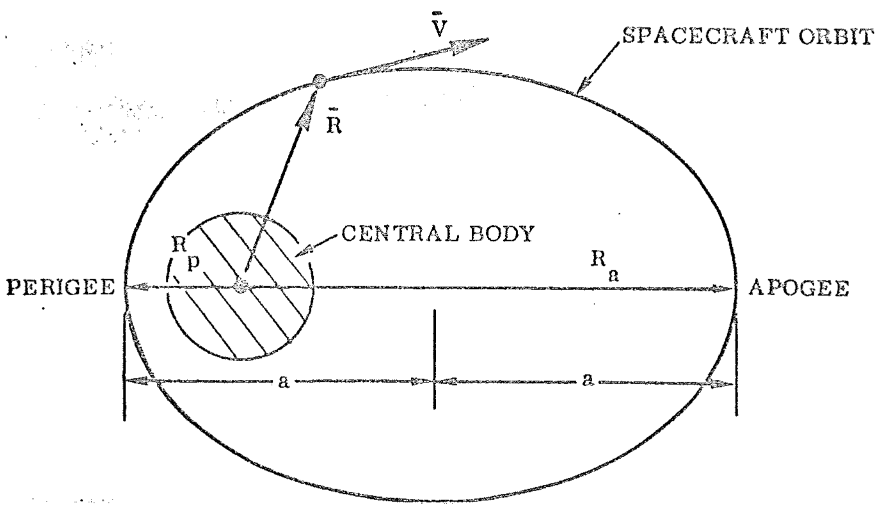
	1	2	3	4	5	6	7	8	9	10	11	12
A	11/ 5/67	78	1911.10	116640.34	0.96968	30.32	285.40	132.94	56.67	92.27	226300.53	11.565
	11/ 5/67	81	1815.64	118649.80	0.97019	29.46	280.41	130.78	56.63	92.56	230319.15	11.865
	11/ 5/67	84	1718.74	120345.23	0.97061	28.86	275.08	129.11	56.67	92.78	233709.80	12.120
	11/ 5/67	87	1627.54	121801.46	0.97096	28.46	269.66	127.99	56.67	92.92	236622.11	12.341
	11/ 5/67	90	1547.74	121930.18	0.97099	28.34	264.02	127.81	56.66	92.99	236879.48	12.360
	11/ 5/67	93	1480.48	122231.58	0.97106	28.43	255.46	128.34	56.67	93.00	237482.28	12.406
	11/ 6/67	78	2083.54	121241.41	0.97084	30.36	284.65	144.36	56.68	91.77	235503.16	12.256
	11/ 6/67	81	1993.48	123232.67	0.97131	29.38	279.76	142.78	56.68	92.12	239485.33	12.559
	11/ 6/67	84	1905.70	124502.30	0.97160	28.89	274.26	141.96	56.65	92.42	242024.29	12.753
	11/ 6/67	87	1822.48	124519.50	0.97160	28.52	268.74	141.44	56.65	92.65	242058.46	12.756
	11/ 6/67	90	1744.96	125687.30	0.97186	28.29	263.27	141.22	56.66	92.81	244393.91	12.936
	11/ 6/67	93	1675.42	124769.53	0.97165	28.39	257.67	141.63	56.68	92.93	242558.24	12.794
	11/ 6/67	96	1613.86	125439.13	0.97180	28.82	252.01	142.68	56.65	92.99	243897.39	12.898
	11/ 6/67	99	1561.42	124703.30	0.97164	29.55	246.45	144.10	56.65	93.01	242425.70	12.784
	11/ 6/67	102	1516.96	124804.09	0.97166	30.42	241.22	145.70	56.68	92.99	242627.30	12.800
	11/ 6/67	105	1478.20	124293.30	0.97154	31.34	236.46	147.26	56.68	92.96	241605.77	12.721
	11/ 6/67	106	1465.66	124199.14	0.97152	31.66	234.98	147.76	56.65	92.95	241417.44	12.707
B	11/ 7/67	78	2256.82	126171.23	0.97198	30.41	283.90	156.21	56.71	91.10	245363.48	13.011
	11/ 7/67	81	2173.60	126409.26	0.97203	29.26	279.08	155.05	56.69	91.50	245839.15	13.047
	11/ 7/67	84	2093.80	127364.26	0.97224	28.83	273.56	154.72	56.68	91.82	247748.82	13.196
	11/ 7/67	87	2016.28	126657.95	0.97208	28.56	267.90	154.65	56.70	92.12	246335.69	13.086
	11/ 7/67	90	1939.90	126958.60	0.97215	28.44	262.28	154.70	56.67	92.38	246936.94	13.133
	11/ 7/67	93	1868.08	127474.95	0.97226	28.47	256.75	154.93	56.67	92.57	247969.45	13.213
	11/ 7/67	96	1801.96	127070.09	0.97217	28.74	251.27	155.46	56.66	92.73	247159.57	13.150
	11/ 7/67	99	1742.68	126849.12	0.97212	29.37	245.85	156.34	56.66	92.82	246717.54	13.116
	11/ 7/67	102	1690.24	126955.39	0.97214	30.30	240.59	157.50	56.66	92.87	246930.04	13.132
	11/ 7/67	105	1644.64	126686.10	0.97208	31.52	235.59	158.81	56.67	92.87	246391.45	13.090
	11/ 7/67	108	1606.74	126278.70	0.97199	32.86	230.92	160.14	56.68	92.86	245576.66	13.027
	11/ 7/67	111	1569.40	125925.38	0.97191	34.31	226.63	161.42	56.68	92.81	244870.06	12.973
	11/ 7/67	114	1538.62	126172.04	0.97197	35.80	222.70	162.61	56.68	92.75	245363.45	13.011
	11/ 7/67	115	1529.50	125933.79	0.97192	36.27	221.47	162.99	56.69	92.72	244866.97	12.974
	11/ 8/67	78	2426.96	127577.13	0.97230	30.34	283.24	167.90	56.75	90.33	248176.04	13.229
	11/ 8/67	81	2353.72	127356.69	0.97225	29.32	278.19	167.40	56.75	90.72	247734.78	13.194
	11/ 8/67	84	2277.34	127838.21	0.97235	28.77	272.80	167.21	56.73	91.09	248697.45	13.269
	11/ 8/67	87	2200.96	127752.47	0.97233	28.51	267.22	167.20	56.71	91.44	248525.61	13.256
	11/ 8/67	90	2125.72	127554.14	0.97228	28.42	261.56	167.33	56.69	91.76	248128.64	13.225
	11/ 8/67	93	2053.90	127737.17	0.97232	28.52	255.93	167.59	56.72	92.02	248494.44	13.254
	11/ 8/67	96	1984.36	128197.28	0.97242	28.61	250.60	168.01	56.67	92.24	249416.43	13.325
	11/ 8/67	99	1921.66	127721.47	0.97231	29.42	245.01	168.59	56.71	92.40	248462.65	13.251
	11/ 8/67	102	1863.52	127607.48	0.97229	30.31	239.83	169.31	56.70	92.52	248234.57	13.233
	11/ 8/67	105	1811.08	127443.52	0.97225	31.47	234.93	170.11	56.70	92.59	247906.57	13.208
	11/ 8/67	108	1764.34	126905.82	0.97213	32.80	230.32	170.95	56.71	92.64	246831.11	13.124
	11/ 8/67	111	1722.16	126517.24	0.97205	34.26	226.04	171.77	56.71	92.64	246053.96	13.064
	11/ 8/67	114	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
	11/ 8/67	115	1633.14	127071.23	0.97217	36.36	220.56	172.78	56.72	92.58	247161.99	13.150
	11/ 9/67	78	2599.96	127274.85	0.97224	30.33	282.52	179.55	56.79	89.53	247572.28	13.182
	11/ 9/67	81	2529.28	127231.87	0.97222	29.36	277.30	179.46	56.78	89.93	247485.92	13.175
	11/ 9/67	84	2456.32	127915.08	0.97237	28.78	272.01	179.47	56.76	90.29	248851.98	13.281
	11/ 9/67	87	2382.22	127264.51	0.97223	28.49	266.48	179.55	56.74	90.68	247550.65	13.180
	11/ 9/67	90	2308.12	127105.87	0.97219	28.40	260.83	179.71	56.74	91.03	247232.82	13.155
	11/ 9/67	93	2235.16	127150.63	0.97220	28.52	255.19	179.93	56.75	91.34	247322.03	13.162
	11/ 9/67	96	2163.34	127564.05	0.97228	28.87	249.64	180.22	56.70	91.63	248148.59	13.227
	11/ 9/67	99	2096.08	127293.52	0.97222	29.44	244.24	180.57	56.70	91.87	247607.29	13.185
	11/ 9/67	102	2033.38	127328.11	0.97223	30.32	239.09	180.95	56.71	92.04	247676.29	13.190
	11/ 9/67	105	1975.24	127141.70	0.97219	31.47	234.24	181.35	56.70	92.18	247303.33	13.161
	11/ 9/67	108	1922.60	126656.02	0.97208	32.79	229.67	181.76	56.72	92.30	246331.87	13.086
	11/ 9/67	111	1874.92	126332.64	0.97201	34.26	225.42	182.16	56.73	92.34	245685.05	13.036
	11/ 9/67	114	1830.46	126017.09	0.97194	35.82	221.49	182.53	56.70	92.37	245053.93	12.987
	11/ 9/67	115	1816.78	126337.97	0.97201	36.35	220.24	182.65	56.69	92.36	245695.69	13.036
	11/10/67	78	2768.68	126190.64	0.97200	30.34	281.82	191.04	56.85	88.78	245404.63	13.014
	11/10/67	81	2701.42	126621.52	0.97210	29.40	276.62	191.27	56.79	89.16	246266.00	13.080
	11/10/67	84	2631.88	127341.54	0.97225	28.82	271.27	191.47	56.78	89.51	247705.69	13.192
	11/10/67	87	2560.06	126278.07	0.97202	28.48	265.73	191.68	56.77	89.91	245578.35	13.027
	11/10/67	90	2487.10	126135.37	0.97198	28.38	260.10	191.88	56.79	90.27	245292.59	13.005
	11/10/67	93	2413.00	126469.06	0.97205	28.52	254.46	192.05	56.78	90.62	245959.62	13.057
	11/10/67	96	2338.90	126616.80	0.97208	28.87	248.91	192.20	56.73	90.96	246254.75	13.080
	11/10/67	99	2268.22	127058.13	0.97218	29.46	243.50	192.34	56.72	91.24	247137.13	13.148
	11/10/67	102	2200.96	126760.70	0.97211	30.33	238.37	192.42	56.72	91.48	246542.04	13.102
	11/10/67	105	2138.26	126068.48	0.97195	31.46	233.56	192.44	56.73	91.68	245157.61	12.995
	11/10/67	108	2080.12	125418.33	0.97181	32.78	229.03	192.46	56.75	91.87	243856.90	12.894
	11/10/67	111	2025.40	125090.21	0.97173	34.24	224.81	192.46	56.72	91.97	243200.56	12.844
	11/10/67	114	1976.38	124879.62	0.97168	35.82	220.88	192.45	56.73	92.05	242779.32	12.811
	11/10/67	115	1960.42	124997.41	0.97171	36.36	219.64	192.45	56.71	92.05	243014.88	12.829
	11/11/67	82	2850.76	125464.73	0.97184	29.17	274.10	203.06	56.86	88.60	243952.97	12.901
	11/11/67	84	2806.30	125656.39	0.97189	28.79	270.49	203.35	56.85	88.82	244336.06	12.931
	11/11/67	87	2735.62	125079.05	0.97175	28.48	265.00	203.65	56.81	89.20	243181.00	12.842
	11/11/67	90	2662.66	125231.25	0.97178	28.39	259.39	203.89	56.79	89.58	243485.04	12.869
	11/11/67	93	2588.56	124829.76	0.97169	28.52	253.75	204.00	56.81	89.94	242641.71	12.804
	11/11/67	96	2512.18	125062.08	0.97174	28.87	248.18	204.02	56.76	90.31	243149.94	12.839
	11/11/67	99	2438.08	125481.43	0.97183	29.50	242.81	203.90	56.76	90.62	243986.33	12.904
	11/11/67	102	2366.24	125079.05	0.97174	30.33	237.65	203.71	56.75	90.92	243179.30	12.842
	11/11/67	105	2297.88	124688.23	0.97169	31.46	232.88	203.36	56.74	91.16	242797.42	12.813
	11/11/67	108	2234.02	123975.23	0.97148	32.78	228.40	202.98	56.75	91.42	240967.16	12.672
	11/11/67	111	2174.74	123742.93	0.97143	34.23	224.21	202.57	56.77	91.56	240506.41	12.637
	11/11/67	114	2118.88	123552.65	0.97138	35.82	220.30	202.16	56.74	91.70	240126.11	12.608
	11/11/67	115	2101.78	123631.80	0.97140	36.36	219.05	202.03	56.76	91.72	240284.01	12.620
	11/12/67	89	2662.16	123943.90	0.97150	28.34	260.40	215.76	56.86	88.87	240911.04	12.666
	11/12/67	90	2637.08	123707.63	0.97144	28.34	258.61	215.82	56.83	89.00	240438.30	12.631
	11/12/67	93	2761.84	123507.10	0.97141	28.50	253.01	215.86	56.83	89.36	240196.95	12.613
	11/12/67	96	2684.32	123515.65	0.97134	28.90	247.50	215.65	56.82	89.72	239653.70	12.571
	11/12/67	99	2605.66	123592.64	0.97141	29.31	242.12	215.32	56.76	90.07	240207.71	12.614
	11/12/67	102	2529.28	123435.08	0.97137	30.37	237.03	214.76	56.79	90.59	239893.65	12.590
	11/12/67	105	2455.18	123241.69	0.97132	31.46	232.22	214.07	56.78	90.67	239505.23	12.560
	11/12/67	108	2384.50	122728.47	0.97120	32.75	227.75	213.27	56.76	90.99	238478.07	12.482
	11/12/67	111	2318.38	122203.93	0.97107	34.22	223.61	212.43	56.75	91.19	237428.78	12.402
	11/12/67	114	2256.82	122054.06	0.97103	35.81	219.73	211.99	56.75	91.38	237128.85	12.379
	11/12/67	115	2237.44	121919.52	0.97100	36.36	218.40	211.32	56.75	91.41	236820.75	12.359

SITE A, 0.83°N LATITUDE AND 24.0°E LONGITUDE. SITE B, 0.42°N LATITUDE AND 1.33°W LONGITUDE.



i	-	ORBITAL INCLINATION
Ω	-	ASCENDING NODE
λ_Ω	-	LONGITUDE OF ASCENDING NODE
ω_p	-	ARGUMENT OF PERIGEE
ω_s	-	ARGUMENT OF POSITION
η	-	TRUE ANOMALY
$\bar{V}$	-	VELOCITY VECTOR
$\bar{\omega}$	-	EARTH ROTATION VECTOR
$\bar{R}$	-	RADIUS VECTOR

Figure 1-10. Orbital Parameters (Sheet 1 of 2)



a - SEMIMAJOR AXIS

e - ECCENTRICITY, $e = (R_a - R_p)/(R_a + R_p)$

R_a - APOGEE RADIUS

R_p - PERIGEE RADIUS

T - ORBITAL PERIOD

Figure 1-10. Orbital Parameters (Sheet 2 of 2) retrothrust applied. The thrust force will be provided by venting hydrogen residuals through the chilldown valve and oxygen residuals through the engine nozzles. The retrothrust duration is programmed for 250 seconds. Based on analytic predictions of tank pressure, thrust, and flow-rate histories, a minimum separation distance of 336 kilometers can be attained at approximately 2.4 hours after injection, assuming minimum tank blowdown impulse. The separation distance for the nominal, maximum, and minimum impulse cases is shown in Figure 1-11.

Figures 1-12 and 1-13 are plots of the predicted thrust and flow-rate histories during retrothrust. The retromaneuver sequence is shown in Table 1-9. Events are referenced from Centaur second main engine start (MES 2).

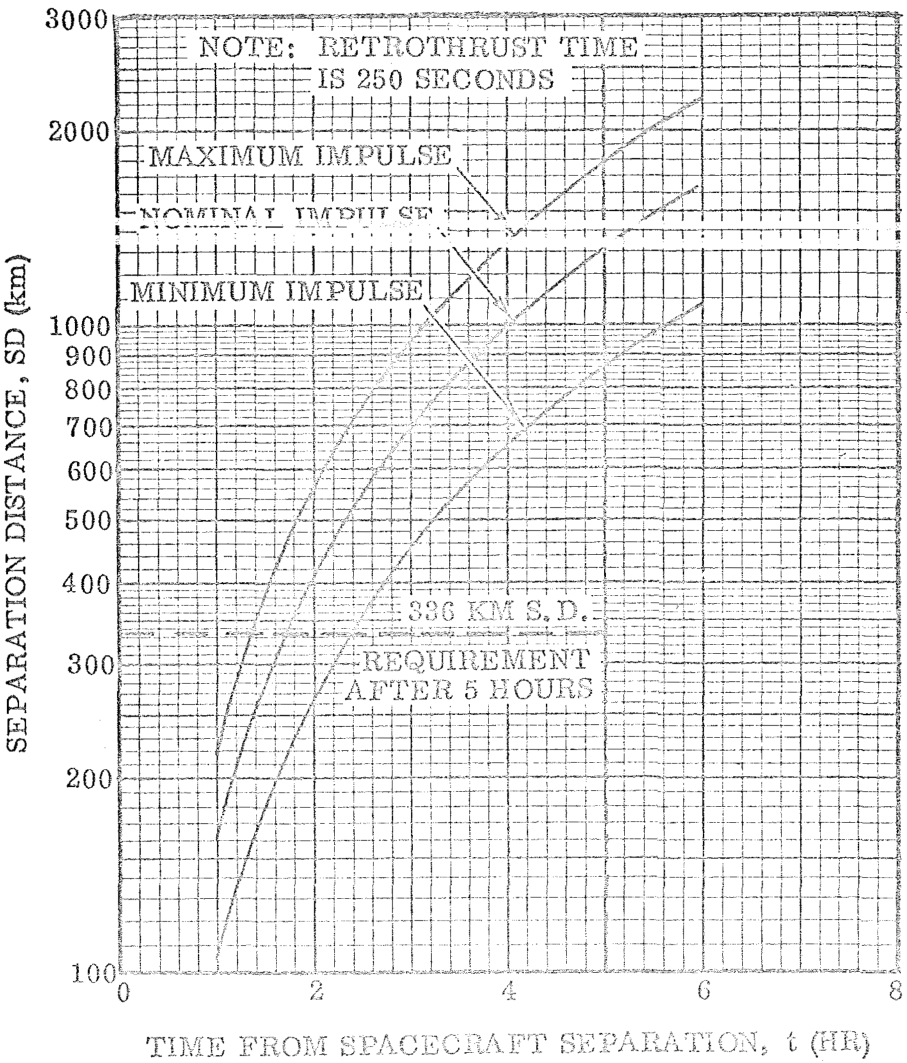


Figure 1-11. Centaur/Spacecraft Separation Distance

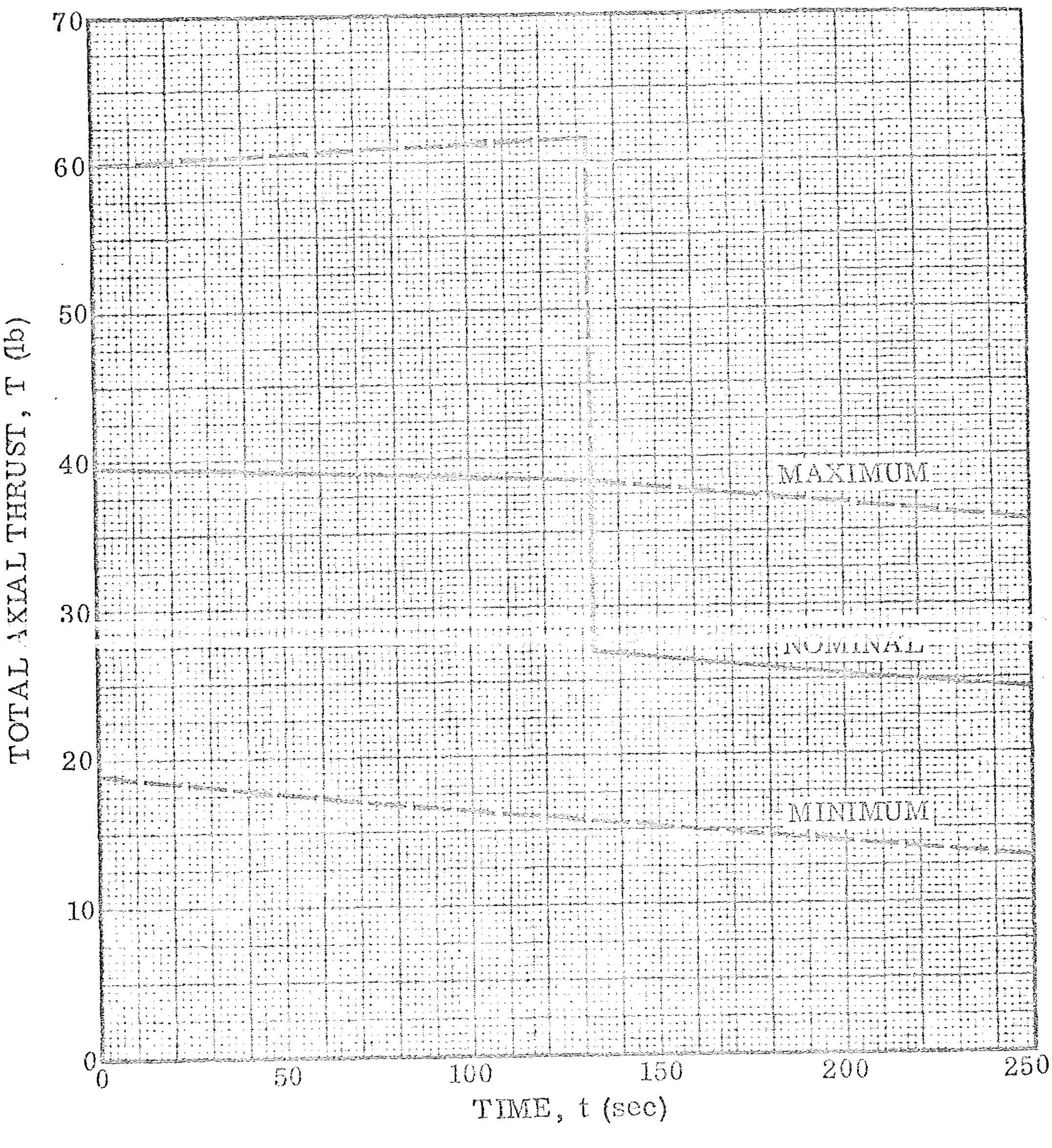


Figure 1-12. Thrust During Centaur Retromaneuver (Tank Blowdown)

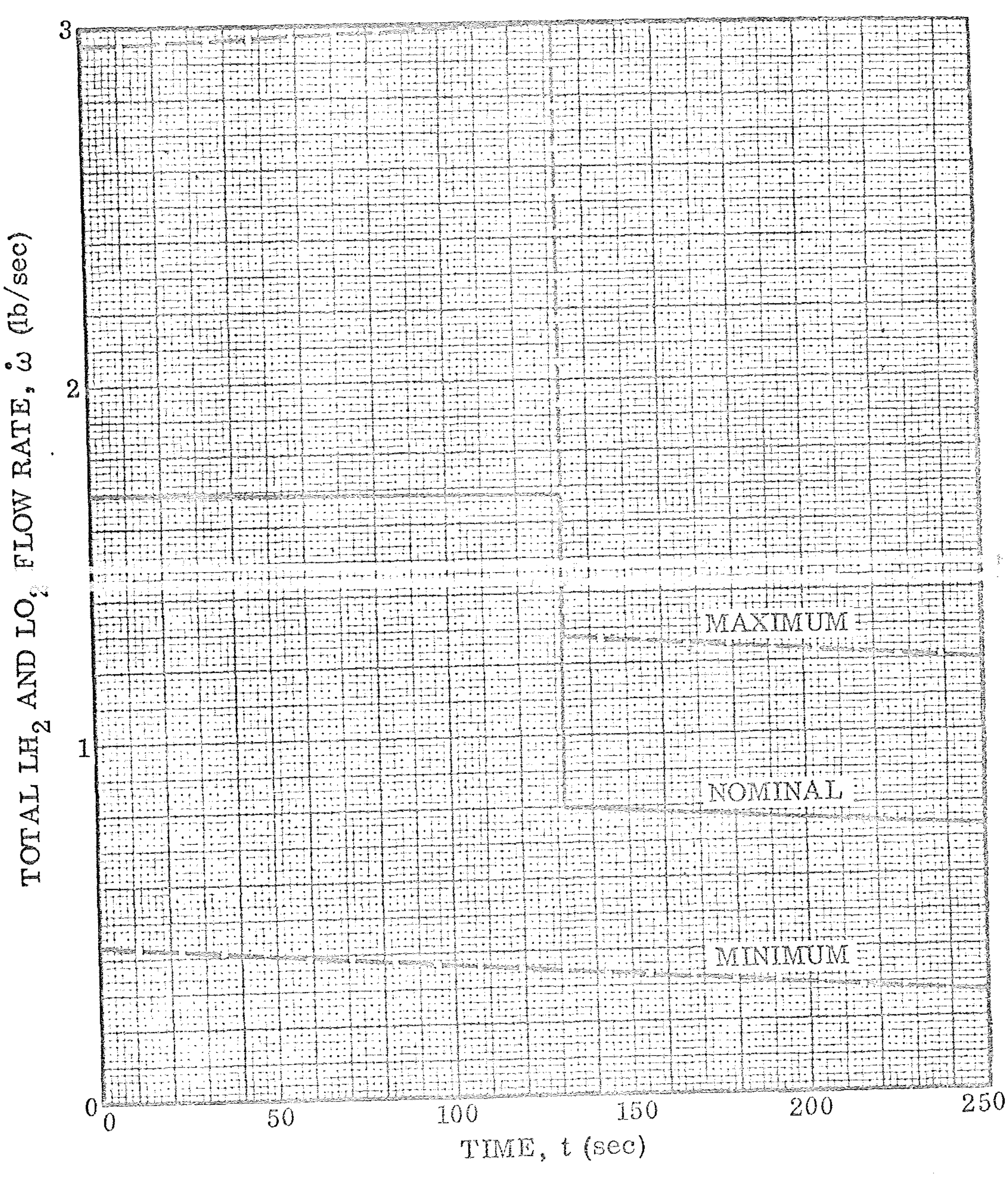


Figure 1-13. Flow Rate During Centaur Retromaneuver (Tank Blowdown)

Table 1-9. Retromaneuver Sequence

EVENT	TIME (sec)	
Separate Spacecraft	MES 2 + 176	t
Start Centaur Turnaround	MES 2 + 181	t + 5
Start Two 50-lb Engines	MES 2 + 221	t + 45
End Two 50-lb Engines	MES 2 + 241	t + 65
Start Tank Blowdown	MES 2 + 416	t + 240
End Tank Blowdown, Restart Two 50-lb Engines	MES 2 + 666	t + 490

The attitude control engines (A and P engines, see Table 8-7) will provide the thrust for Centaur turnaround. During Centaur turnaround the two 50-pound thrust engines (V engines) will provide the initial impulse for Centaur/Spacecraft separation distance to preclude impingment of the tank blowdown exhaust on the spacecraft. Limit rate during the turnaround is 1.6 degrees per second about either the pitch or yaw axis, yielding a maximum rate of 2.25 degrees per second. First firing, of the 50-pound rockets during the retromaneuver sequence, is timed to coincide with the approximate 90-degree vehicle attitude with respect to the velocity vector. Following Centaur turnaround, tank blowdown will be initiated and will provide the major portion of the impulse necessary for the required Centaur/Spacecraft separation distance. After completion of the Centaur retromaneuver the vernier half on phase is re-initiated and continues until H_2O_2 depletion.

SECTION 2

PERFORMANCE

This section defines vehicle performance in terms of excess propellants and payload capability. To aid the assessment of configuration changes, performance exchange data are also presented.

2.1 EXCESS PROPELLANTS

For firing window determination purposes, excess propellant is defined as the total weight of unburned propellants remaining after second main engine cutoff (MECO 2) and thrust decay. The variation of excess propellants during the firing window is determined from the targeting procedure described in Section 1.3. Daily variations for the November 1967 launch opportunity are reported in Reference 2.

To ensure successful completion of the mission, a performance reserve is set aside to account for non-nominal vehicle performance and contingencies. Therefore, excess propellants can be no less than the specified performance reserve (see Section 2.3) at any time during the launch window. Based on a Centaur jettison weight of 4111 pounds and a spacecraft weight of 2223 pounds, the lowest excess propellants for the November launch opportunity is 615 pounds (includes 255-pound performance reserve).

2.2 PAYLOAD CAPABILITY

Vehicle payload capability is a parameter used for the purpose of monitoring configuration and performance changes (Reference 4). The Payload capability is obtained by solving the following equation.

$$\text{Payload capability} = WBO - PR - WJETT \qquad (2-1)$$

where,

WBO	=	Burnout weight, total weight injected into the required terminal orbit, assuming nominal functioning of all systems and consumption of all propellants available for main impulse (excluding performance reserve)
PR	=	Performance Reserve, amount of Centaur stage propellants held in reserve
WJETT	=	Centaur stage jettison weight after main engine thrust decay (Centaur tank weight at separation)

Payload capability, as defined in this section, is based on a set of reference conditions that define a single trajectory that is used for vehicle payload capability analysis on a bi-monthly basis (Reference 4). The payload capability is based on the reference conditions presented in Table 2-1.

Table 2-1. Reference Conditions

1.	ETR Launch Site	Complex 36B
2.	Launch Azimuth	114 deg
3.	Parking Orbit Altitude	90 n. mi.
4.	Injection Energy	$-0.85\ km^2/sec^2$
5.	Injection True Anomaly (MECO 2)	4 deg
6.	Parking Orbit Coast Time	25 min.

2.3 PERFORMANCE RESERVE

2.3.1 DEFINITION OF FLIGHT PERFORMANCE RESERVE

The flight performance reserve (FPR) is the propellant held in reserve to ensure achievement of the required mission velocity under non-nominal (but normal) flight conditions. FPR is computed using a Monte Carlo method to estimate the effects of a number of vehicle subsystem performance dispersions on net vehicle performance and propellant margin. Details of the Monte Carlo FPR Program and computation of FPR are documented in Reference 5.

The flight performance reserve propellants remain unused providing all vehicle subsystems function according to the predicted mean on a normal distribution curve; a condition reflecting nominal performance. Although, normally, FPR is considered as a propellant reserve that could be used up in flight, it should be recognized that the probability of better than nominal performance is equally likely. In this event, the propellant remaining on board at termination of the powered flight phase could be as much as twice the FPR value. Only under an adverse statistical combination of performance deviations would no usable propellants remain on board. In reference to the payload capability equation, Equation 2-1, which indicates one-to-one relationship between payload capability and FPR, the "guarantee" of a given payload capability via assignment of FPR is illustrated in Figure 2-1.

The Atlas/Centaur payload capability is based on the requirement of ensuring achievement of the mission orbital energy in the presence of a 3-sigma statistical combination of performance deviations. This results in a 99.87 percent probability of successfully completing the mission. It should be noted that the area under the normal distribution curve, Figure 2-1, is truncated only at the guaranteed payload (3-sigma). This yields a success probability that is somewhat higher than the conventional definition of probability associated with a ±3-sigma bounded normal distribution curve (in which the probability would be 99.73 percent).

For the Atlas/Centaur missions, the propellant reserve is provided only in the Centaur stage. The Atlas stage completion is determined by propellant depletion regardless of velocity achieved during Atlas phase of flight. Thus, performance deviations of all stages are compensated for by the propellant reserve of the Centaur stage. This results in the most efficient utilization of booster vehicle capability.

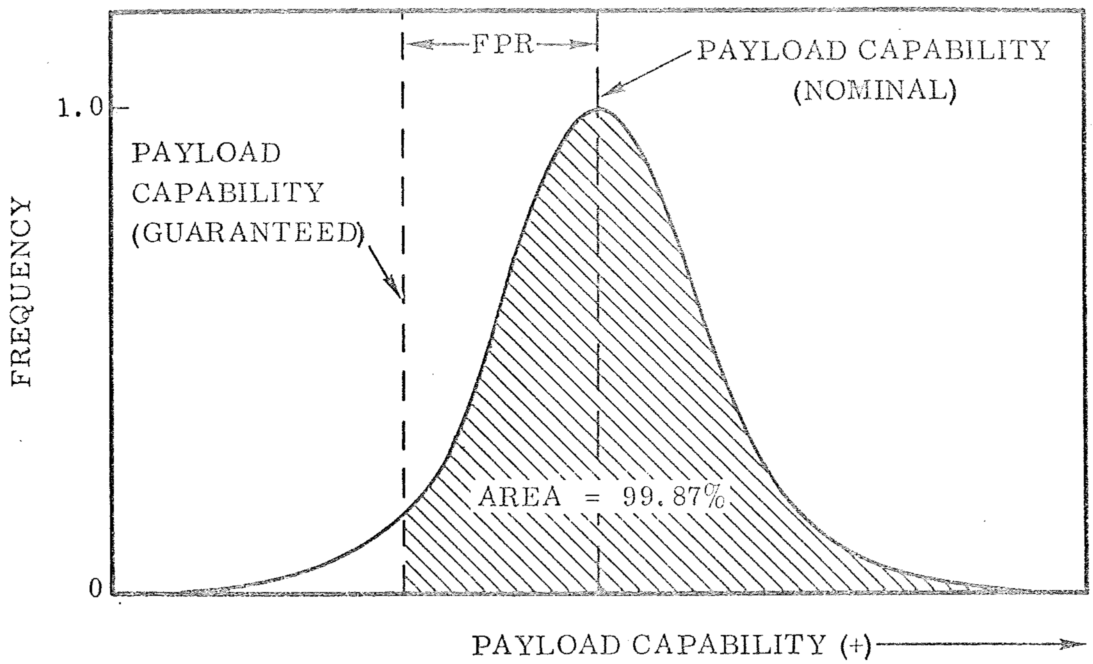


Figure 2-1. Payload Capability/FPR Relationship

2.3.2 NET 3-SIGMA FLIGHT PERFORMANCE RESERVE

In addition to the normal performance deviations that give rise to the FPR requirement, any apparent uncertainty such as the aerodynamic drag model is algebraically added to the FPR as a bias.

The drag model has been recently revised to reflect experience gained from Atlas and LV-3C Atlas/Centaur flights. The revision reduces the predicted drag on the vehicle and yields increased performance capability. Pending further flight experience, a 30-pound FPR bias is being provided to increase the confidence level in predicted performance when the revised drag model is used in trajectory calculations.

An additional 60-pound contingency is currently provided to further increase the confidence level in predicted performance.

Table 2-2 presents the net 3-sigma flight performance reserve, that is, FPR plus drag model bias plus contingency.

Table 2-2. Net 3-Sigma Flight Performance Reserve

FPR	165 lb
Drag Model Bias	30 lb
Contingency	60 lb
Net 3-Sigma FPR	255 lb

2.4 PERFORMANCE EXCHANGE COEFFICIENTS

This section presents data that relate excess propellant to various independent parameters such as vehicle weights, propulsion characteristics, propellants, and mission parameters. The linear relationships are shown as exchange coefficients, partial derivatives of excess propellants with respect to the independent variable (see Figure 2-2). These coefficients are related to the excess propellant increment as defined by the equation

$$\Delta WEP = \sum_{i=1}^{n} \frac{\partial WEP}{\partial x_i} \times \Delta x_i \qquad (2\text{-}2)$$

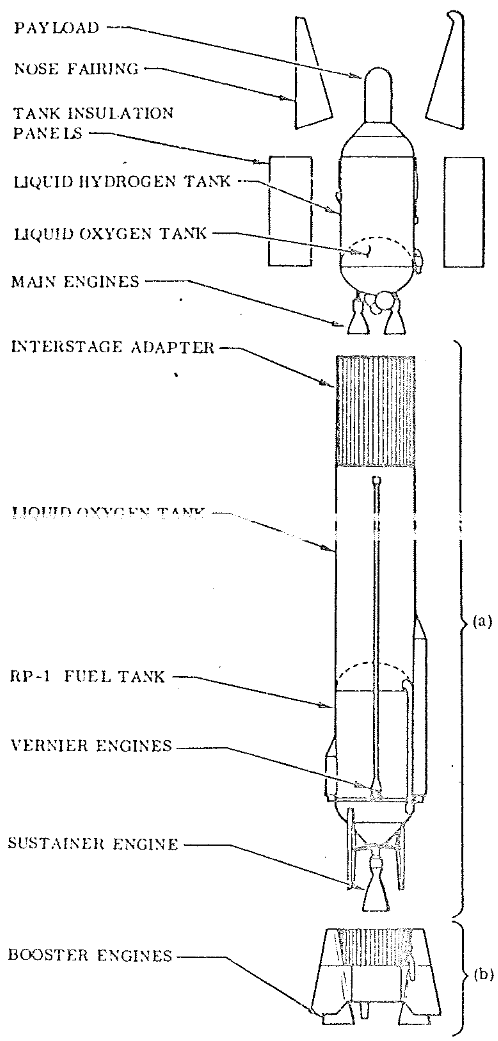
where

ΔWEP = Net excess propellant increment

$\frac{\Delta WEP}{\partial x_i}$ = Excess propellant exchange coefficient with respect to the i^{th} independent variable (x_i)

Δx_i = Increment in the i^{th} independent variable

It should be noted that all performance exchange coefficients were computed via closed-loop detailed trajectory simulations. The AC-13 closed-loop guidance simulation (see June issue of Reference 4) was used for the computation. Because of the similarity between the vehicles, the AC-13 Performance Exchange Coefficients are applicable to the AC-14 mission.



(a) SUSTAINER PACKAGE
(b) BOOSTER PACKAGE

	INDEPENDENT VARIABLE	EXCHANGE COEFFICIENT	UNITS
I	CENTAUR STAGE		
	Stage Inert Weight	-1.0	lb of excess propellants per lb
	Nose Fairing	$\frac{1}{-10.6}$	lb of excess propellants per lb
	Insulation Panels	$\frac{1}{-12.3}$	lb of excess propellants per lb
	Thrust	$\frac{1}{67.3}$	lb of excess propellants per lb
	Impulse Propellants	$\frac{1}{13.7}$	lb of excess propellants per lb
	Specific Impulse	$\frac{1}{25.9}$	lb of excess propellants per sec
	A/C Separation Coast Time	-3.9	lb of excess propellants per sec
	Boost Pump H_2O_2	$-171.0^{(1)}$	lb of excess propellants per lb/sec
II	ATLAS STAGE		
	Sustainer Package (a)	$\frac{1}{-10.3}$	lb of excess propellants per lb
	Sustainer Thrust	$\frac{1}{-106}$	lb of excess propellants per lb
	Booster Thrust	$\frac{1}{422}$	lb of excess propellants per lb
	Booster Package (b)	$\frac{1}{-10}$	lb of excess propellants per lb
	Impulse Propellants		
	Fuel Wt. Change >0	$\frac{1}{70.6}$	lb of excess propellants per lb
	<0	$\frac{1}{-48.9}$	lb of excess propellants per lb
	Oxidizer Wt. Change >0	$\frac{1}{100.0}$	lb of excess propellants per lb
	<0	$\frac{1}{-63.7}$	lb of excess propellants per lb
	Sustainer Specific Impulse	17.2	lb of excess propellants per sec
	Booster Specific Impulse	25.1	lb of excess propellants per sec
III	MISSION PARAMETERS		
	Injection Energy	-70.0	lb of excess propellants per km^2/sec^2

(1) During Main Engine Operation

Figure 2-2. Performance Exchange Coefficients

SECTION 3

RANGE SAFETY

For each Atlas/Centaur flight, range safety and range planning trajectory data are prepared. The data prepared are stored on IBM magnetic tapes and are presented in various reports as a data package.

3.1 DATA PACKAGE

The data package for the AC-14 flight consists of

a. Nominal and dispersed trajectory data, Reference 6, composed of

1. Planned nominal trajectory data.
2. Dispersion trajectory data in the form of ΔX, ΔY, ΔZ, necessary for defining present position boundaries, and $\Delta \dot{X}$, $\Delta \dot{Y}$, $\Delta \dot{Z}$ used for defining the instantaneous impact point (IIP) envelopes.

b. Re-entry data composed of

1. Expected impact points (IP) for the Atlas booster package, Atlas sustainer stage, Centaur insulation panels, and Centaur nose fairing.
2. Dispersion envelopes of impact points for each of the above items.
3. Expected effects of destruct action including the number and approximate weights of resulting fragments, estimates of drag coefficient histories for major fragments, and velocity increments imparted to these fragments by destruct action (explosion).

c. Velocity vector turn data, Reference 8, composed of velocity vector turning capability data assuming vacuum tumbling turns caused by engine lockup at various constant deflection angles.

d. Impact probability data, Reference 9, composed of

1. Impact probabilities for points on land masses.

2. Kill probabilities for flights over land masses in terms of the probability of killing a single person.

SECTION 4

TRACKING DATA

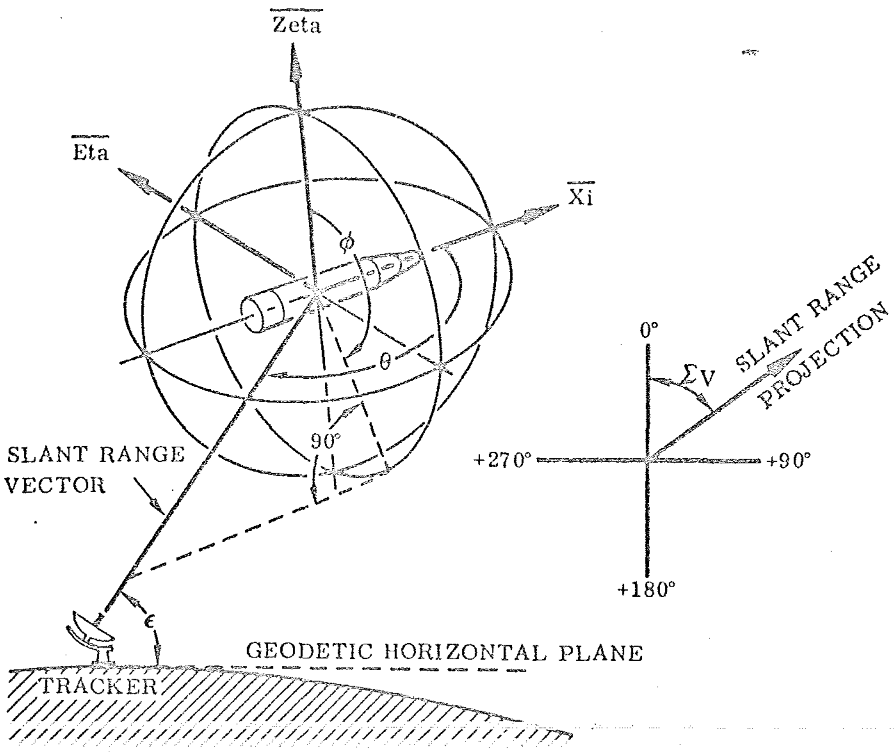
Vehicle position, velocity, and acceleration are calculated from tracking data. This information is used for range support during the flight and for postflight performance analysis. The quality of the tracking data, and hence the adequacy of the range support and performance analysis, is dependent on how well the ground tracking stations can monitor the trajectory during the flight. Down-range ground stations, with precisely defined coordinates, have been established for this purpose.

A list of expected tracking stations and their coordinates are presented in Table 4-1. The coordinates are referenced to the Clark spheriod of 1866. Figure 4-1 defines the tracking parameters used during tracking.

Table 4-1. Tracking Station Coordinate Data*

STATION	GEODETIC LATITUDE (deg)	WEST LONGITUDE (deg)	HEIGHT ABOVE REF. SPHERIOD (ft)	TYPE
1. Cape Kennedy Tel IV	28.463667	80.653028	59.7	Telemetry
2. Merritt Island 19.18	28.424424	80.664618	48.6	Radar
3. Grand Bahama 3.18	26.635837	78.267964	50.9	Rad/Tel
4. Grand Turk	21.443503	71.147931	22.9	Rad/Tel
5. Bermuda	32.254252	64.839201	259.0	Rad/Tel
6. Antigua 91.18	17.143055	61.793377	182.5	Rad/Tel
7. Ascension 12.18	-7.975147	14.402707	673.4	Rad/Tel
8. Pretoria 13.16	-25.947274	-28.356704	5454.2	Rad/Tel
9. Tananarive	-18.800000	-47.500000	100.0	Telemetry
10. Grand Canary Island	27.735522	15.600005	55.7	Rad/Tel
11. Carnarvon	-24.900000	-113.716670	164.0	Rad/Tel

* The coordinates are referenced to the Clark spheriod of 1866 and were obtained from Reference 10.



DEFINITIONS

1. LOOK PHI (ϕ) - The angle between the pitch plane (xi-zeta) and the projection of the slant range vector on the base of the vehicle. It is measured positive clockwise from the positive zeta (yaw) axis, when viewing along the positive direction of the longitudinal ($\overline{xi}$) axis.

2. LOOK THETA (θ) - The angle between the longitudinal ($\overline{xi}$) axis of the vehicle and the slant range vector from vehicle to tracker, measured from the positive $\overline{xi}$ axis.

3. ELEVATION ANGLE (ϵ) - The angle of the slant range vector with respect to the geodetic horizontal plane at the tracker.

4. VIEWING AZIMUTH (Σ_V) - The azimuth of the slant range vector measured in the geodetic horizontal plane at the tracker.

5. SLANT RANGE - Vehicle distance from tracker in nautical mile.

Figure 4-1. Tracking Parameter Definitions

Parametric tracking data for the following stations are presented in Figures 4-2 through 4-6.

a. Cape Kennedy Tel. II

b. Bermuda

c. Grand Turk

d. Antigua 91.98

e. Trinidad

f. Tananarive

g. Pretoria

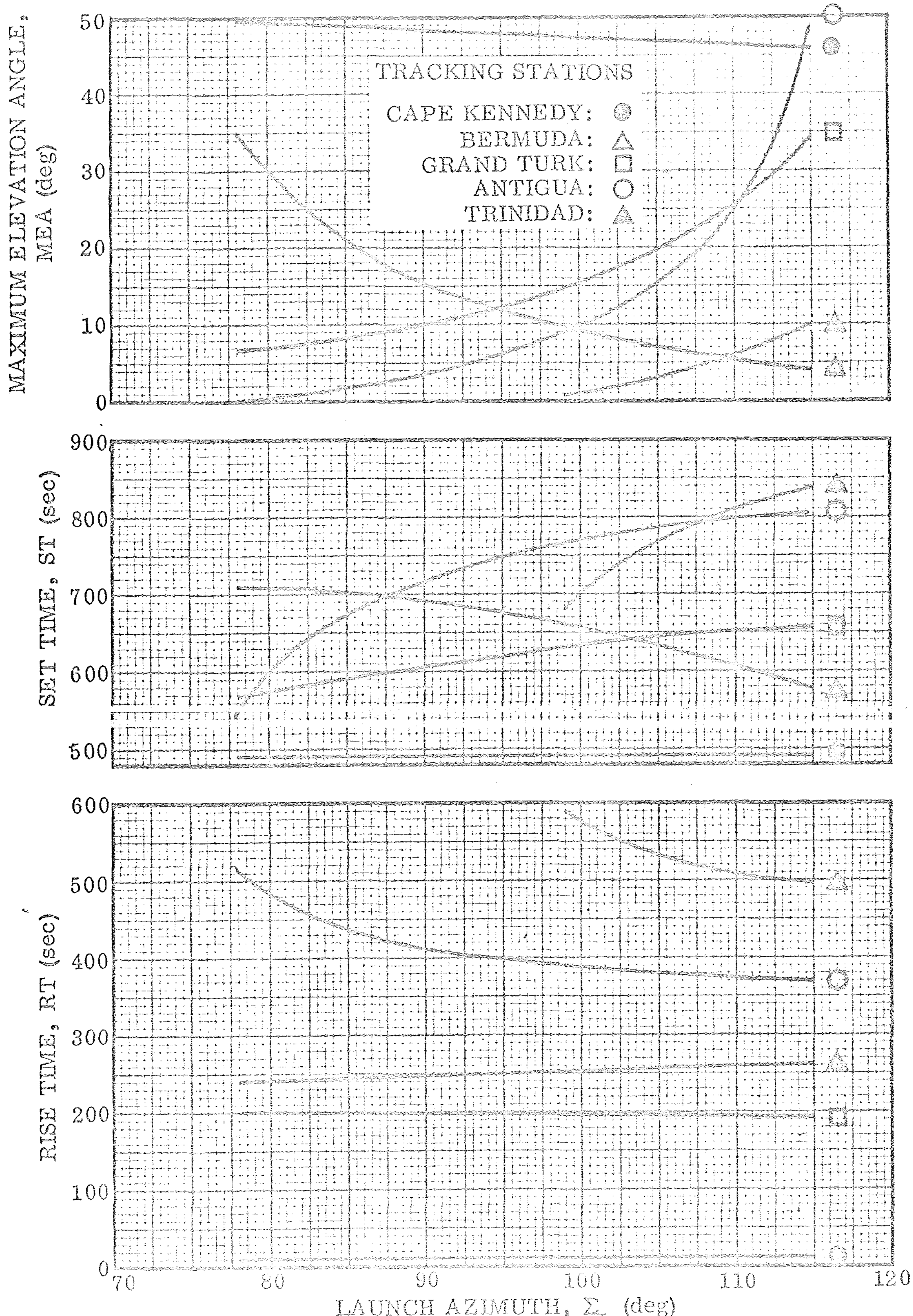
h. Carnarvon

i. Grand Canary Island

j. Ascension

Figure 4-2 gives rise time, set time, and maximum elevation angle as a function of launch azimuth for the first five tracking stations. Data for these stations are independent of parking orbit coast time. Figures 4-3 through 4-6 give rise time as a function of parking orbit coast time for the remaining tracking stations. Data for these stations are a function of parking orbit coast time. The AC-14 Firing Tables report, Reference 2, contains launch azimuth and parking orbit coast-time/launch-time relationships.

It should be noted that the AC-14 tracking data were based on range support data from Reference 10. These data should be applicable for all remaining Atlas/Centaur/Surveyor parking orbit ascent missions.



NOTE: 1. ELEVATION ANGLE IS APPROXIMATELY 1 DEGREE FOR RT AND ST.
2. RT AND ST ARE REFERENCED TO LAUNCH VEHICLE 2-INCH MOTION.

Figure 4-2. Tracking Station Parameter Data

(To be published later)

Figure 4-3. Grand Canary Island and Ascension Tracking Station Parameter Data

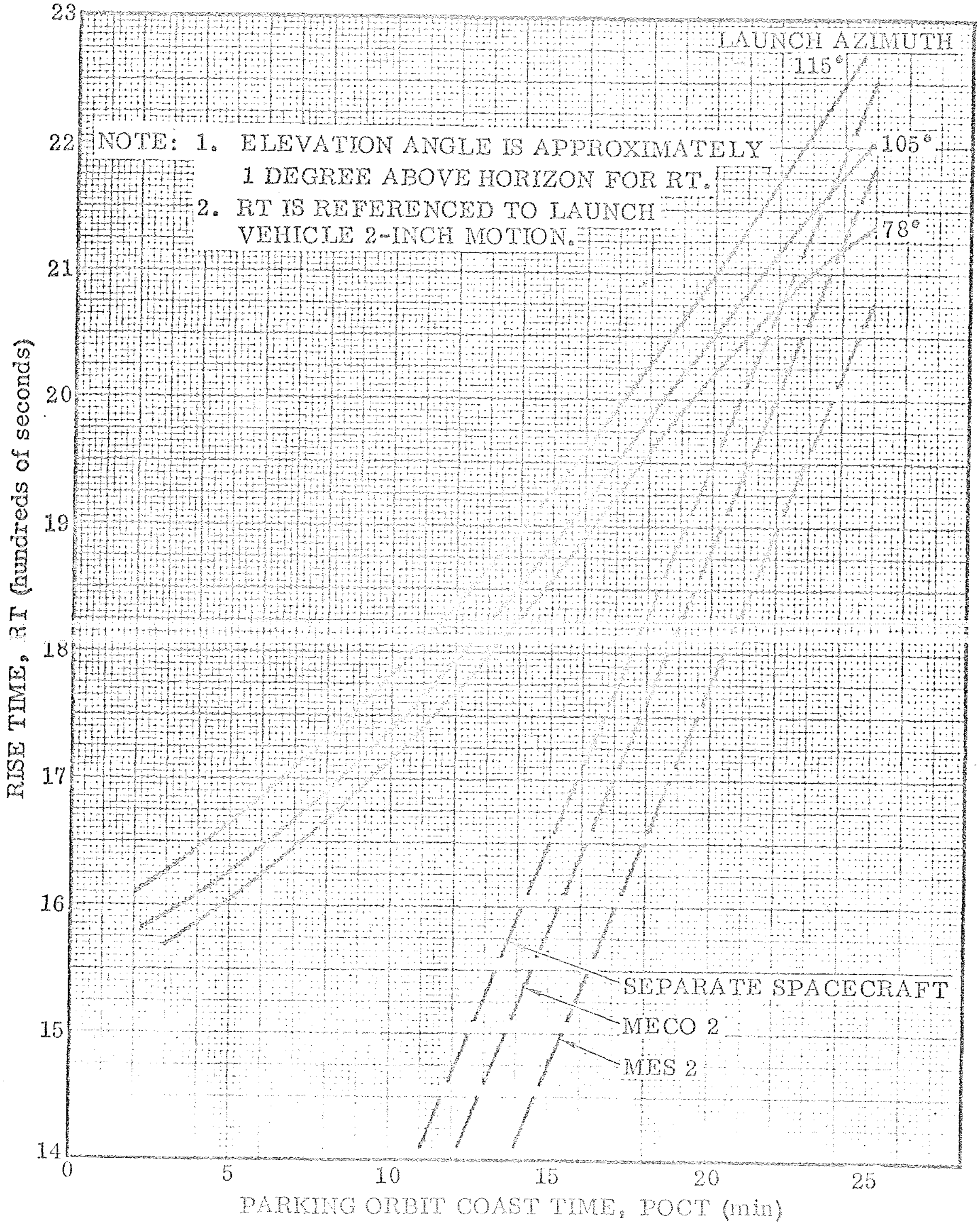


Figure 4-4. Tananarive Tracking Station Parameter Data

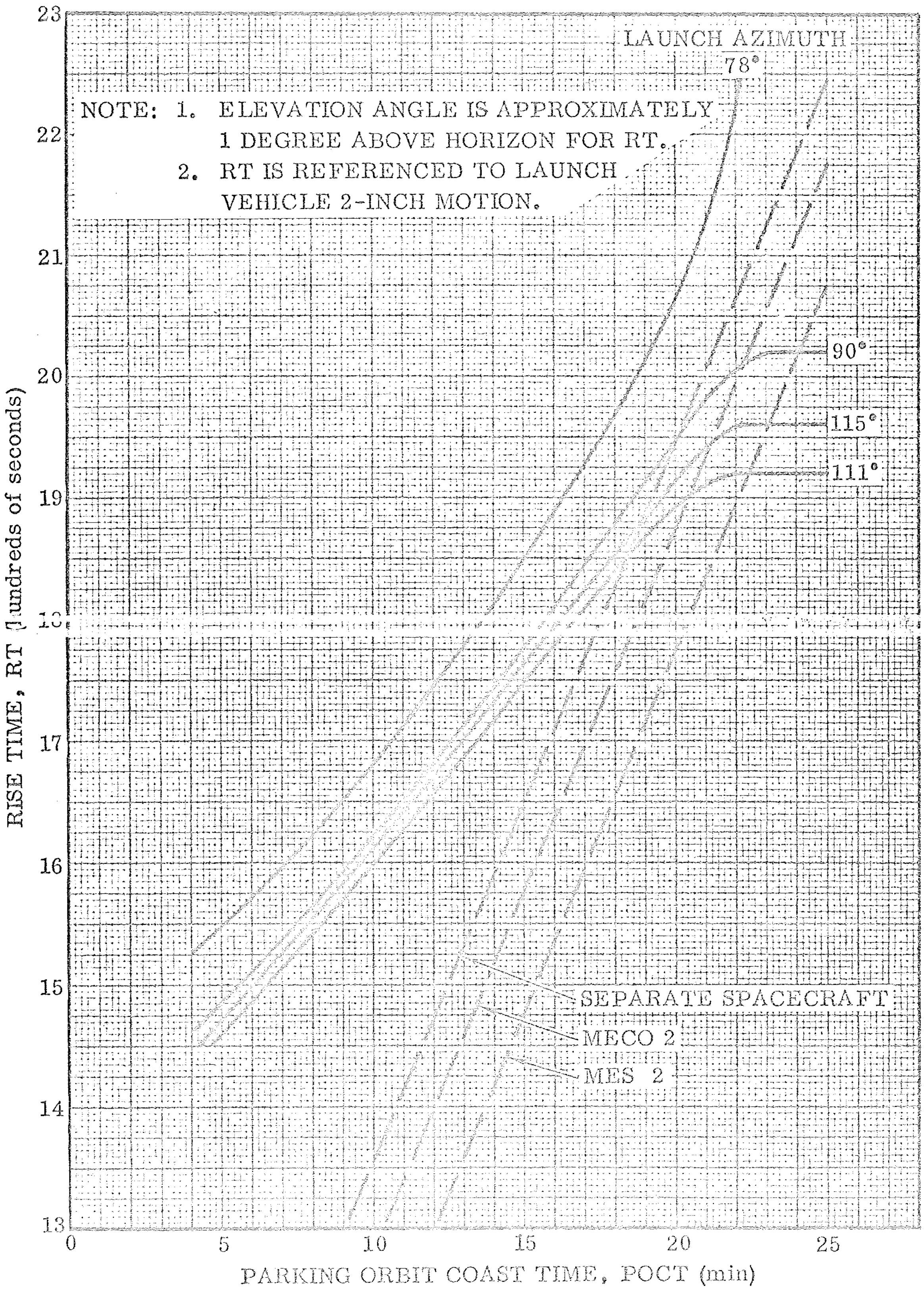


Figure 4-5. Pretoria Tracking Station Parameter Data

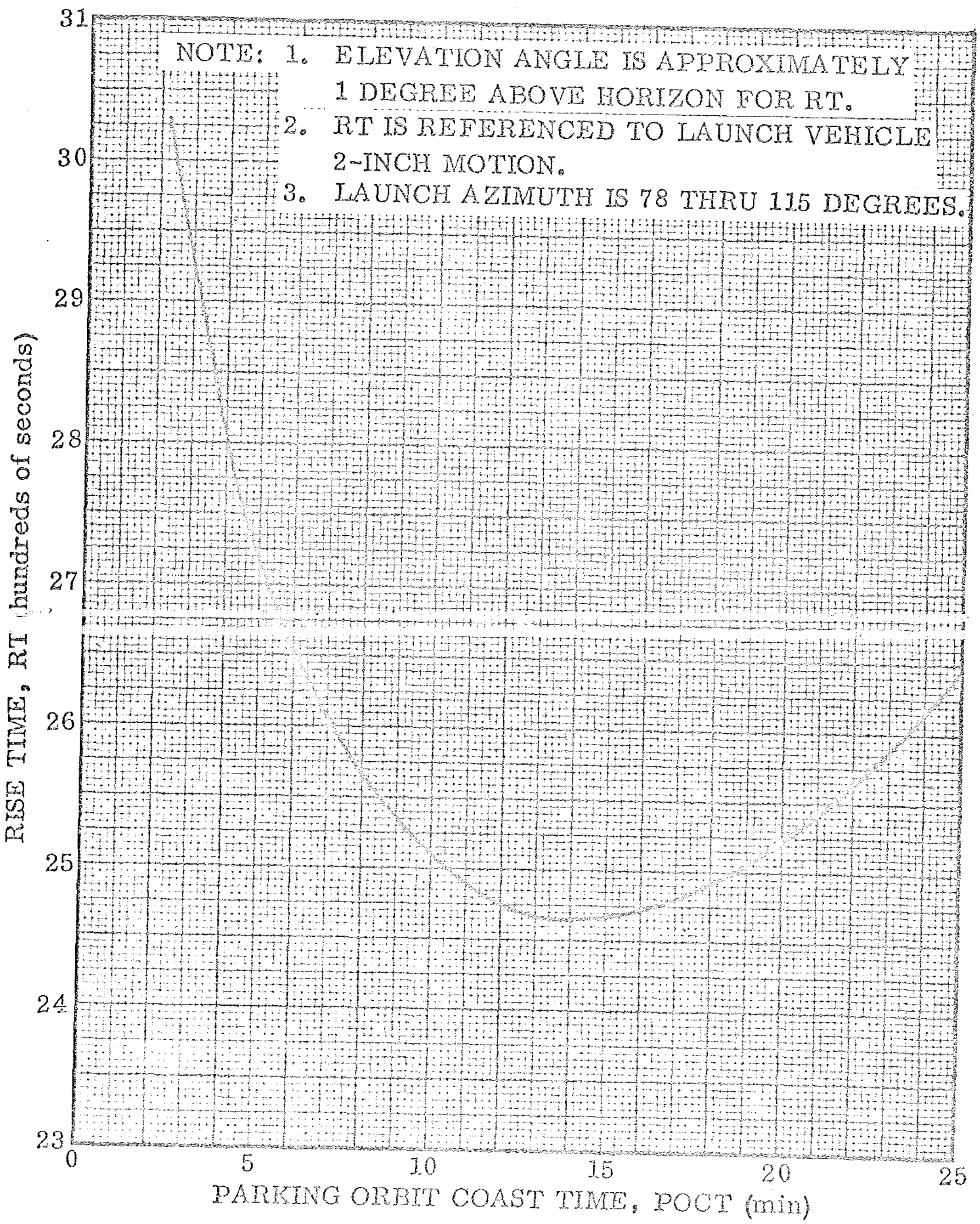


Figure 4-6. Carnarvon Tracking Station Parameter Data

SECTION 5
FLIGHT MECHANICS

This section introduces some of the flight mechanic concepts of earth-moon trajectory design. The topics discussed are 1) earth-moon geometric properties, 2) pre-injection trajectory phase, 3) post-injection trajectory phase, and 4) launch window.

5.1 EARTH-MOON GEOMETRIC PROPERTIES

The complex geometric problem associated with earth-moon trajectory design can be visualized by first considering the orbital motion of various bodies.

a. The earth rotates about its polar axis with an angular rate of approximately 15 degrees per hour. Its equatorial plane is inclined to the ecliptic (earth's orbit plane about the sun) by about 23.5 degrees.

b. The moon orbits the earth with a sidereal period of about 27.3 days. This rotation of the moon about the earth causes day-to-day variations in its right ascension (angular measure eastward along the earth equator from the vernal equinox).

c. Since the moon rotates about its own axis once every 27.3 days, it presents the same surface area to an observer stationed on the earth.

d. The moon's orbit is slightly eccentric, 0.05; therefore the earth-moon radial distance varies as a sinusoidal function.

e. Because the plane of the lunar orbit is inclined to the ecliptic approximately 5 degrees, declination of the moon is a sinusoidal function with a period equal to the lunar month. Regression of the nodes of the lunar orbit causes the amplitude of lunar declination to vary from about plus and minus 18.5 to 28.5 degrees. The time between minimum and maximum amplitude is about 9.3 years.

Figures 5-1 through 5-5 present in graphical form some of the earth-moon geometric properties discussed above.

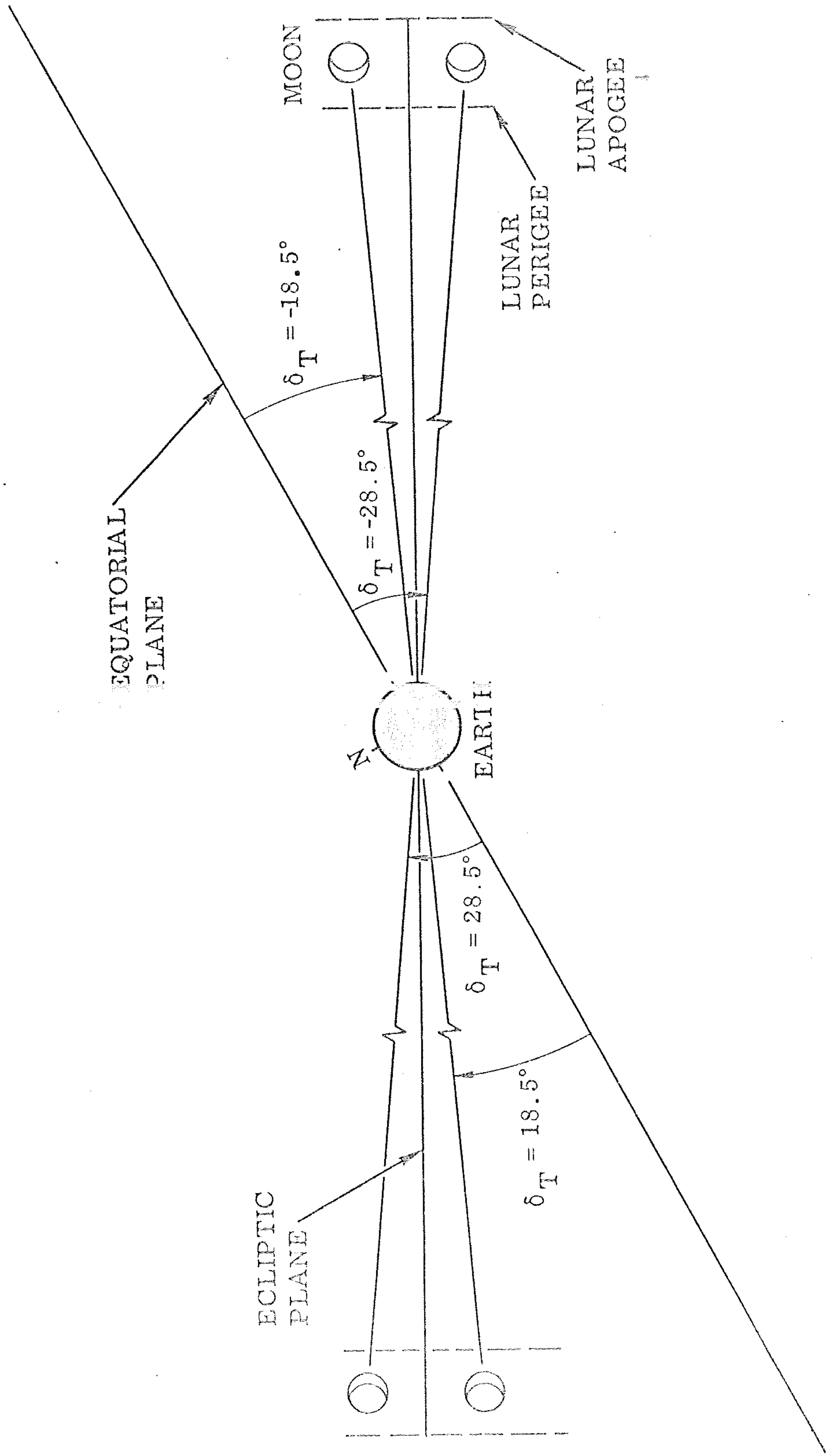


Figure 5-1. Earth-Moon Geometric Relationship

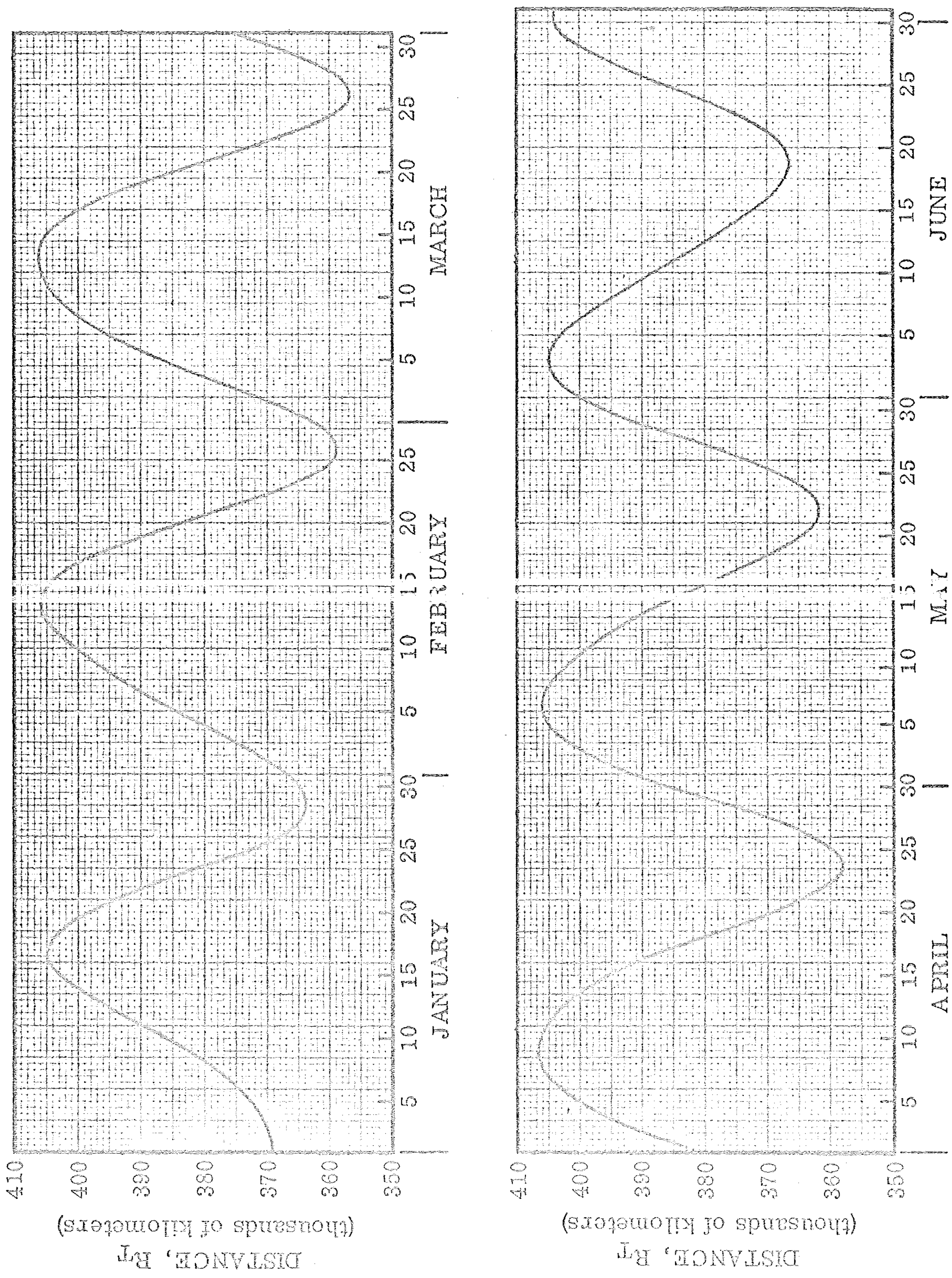


Figure 5-2. Earth-Moon Distance, 1967 (Sheet 1 of 2)

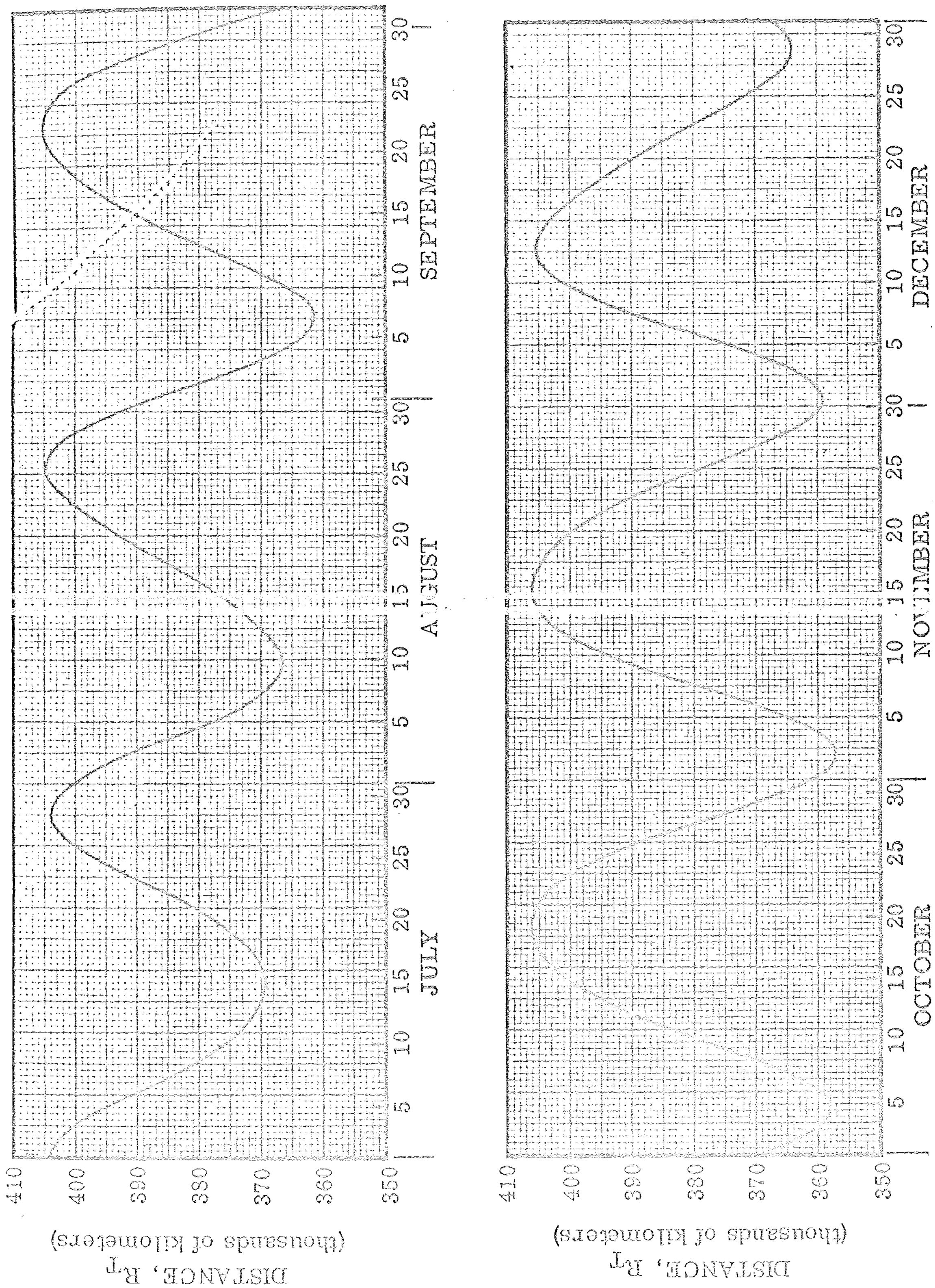


Figure 5-2. Earth-Moon Distance, 1967 (Sheet 2 of 2)

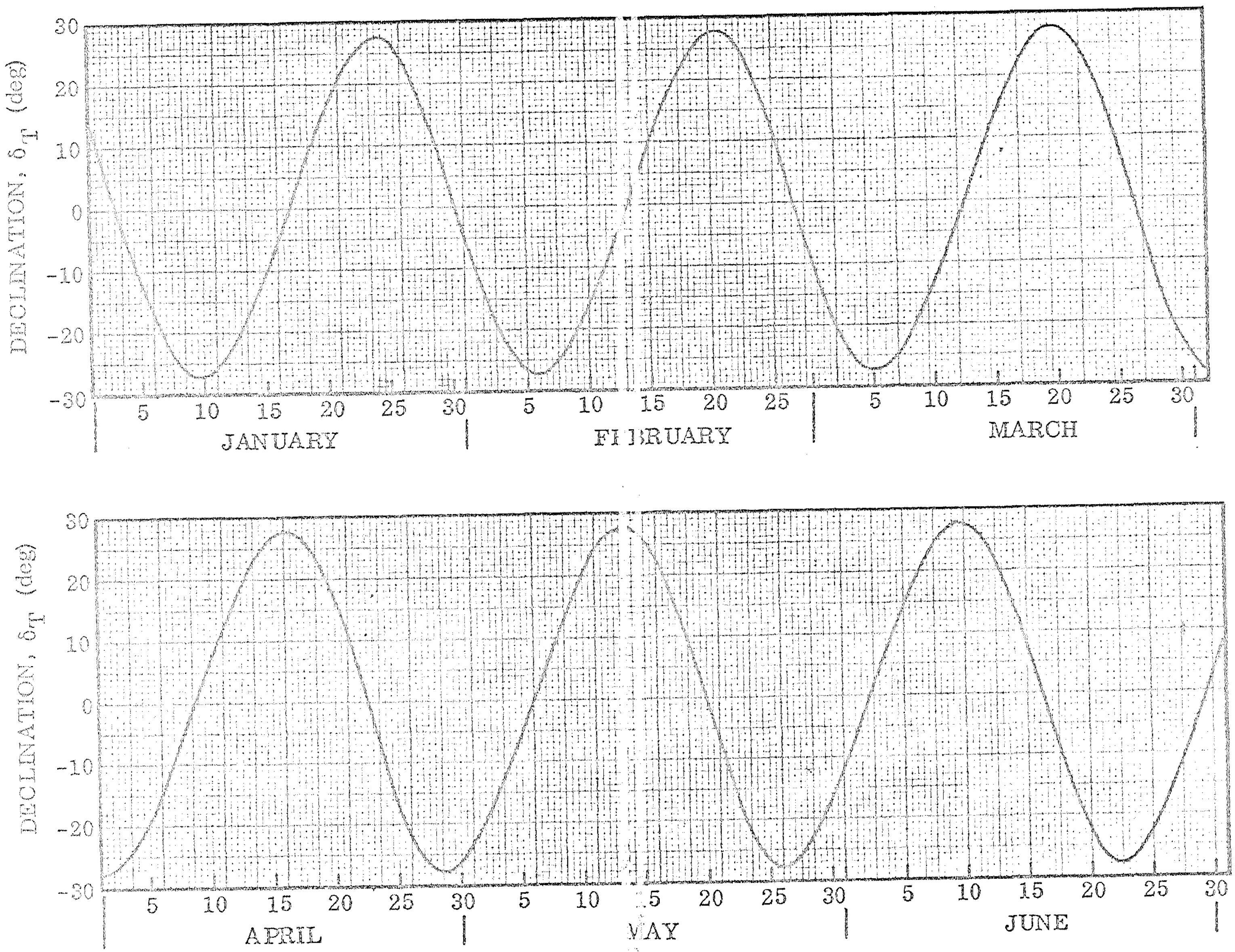


Figure 5-3. Lunar Declination, 1967 (Sheet 1 of 2)

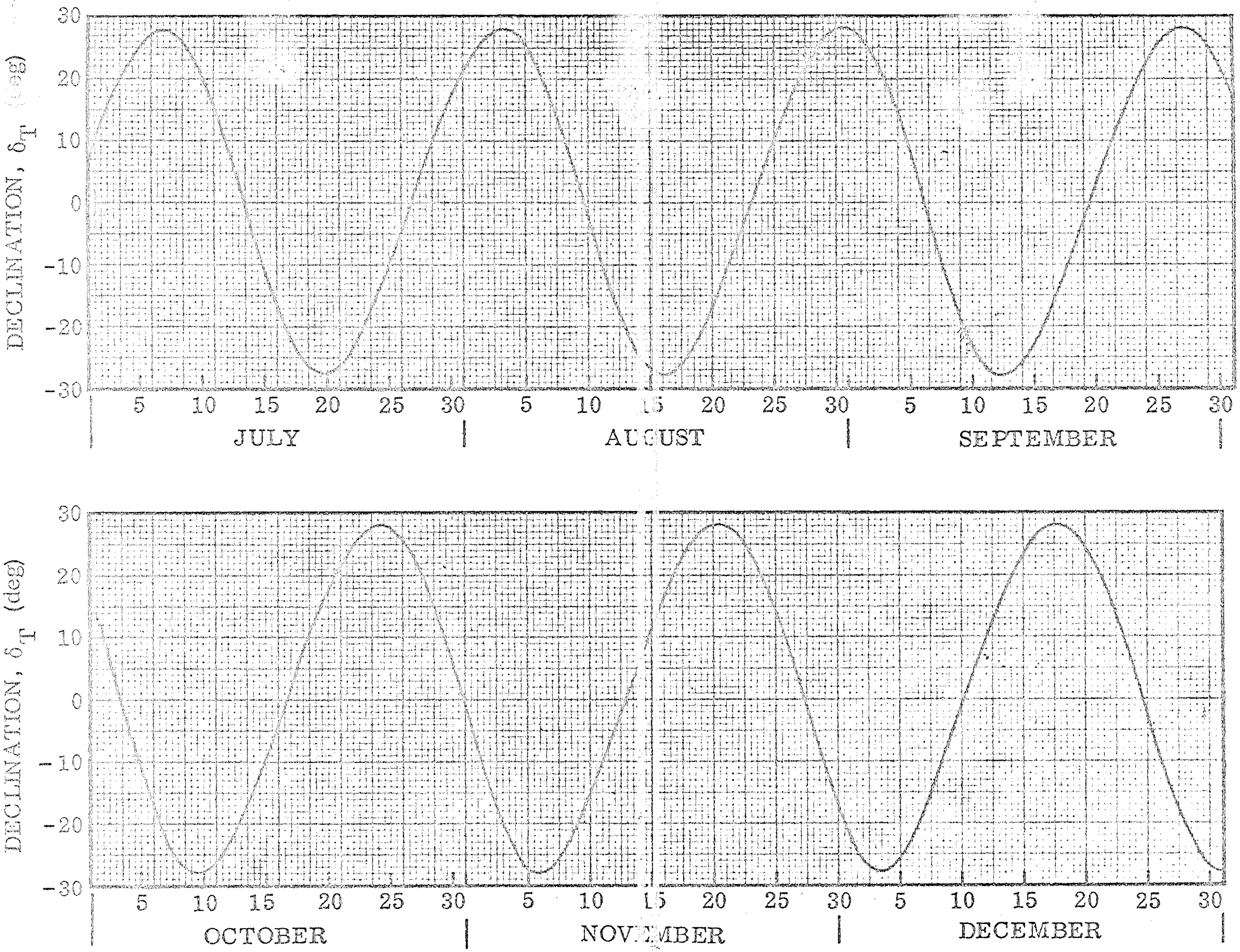


Figure 5-3. Lunar Declination, 1967 (Sheet 2 of 2)

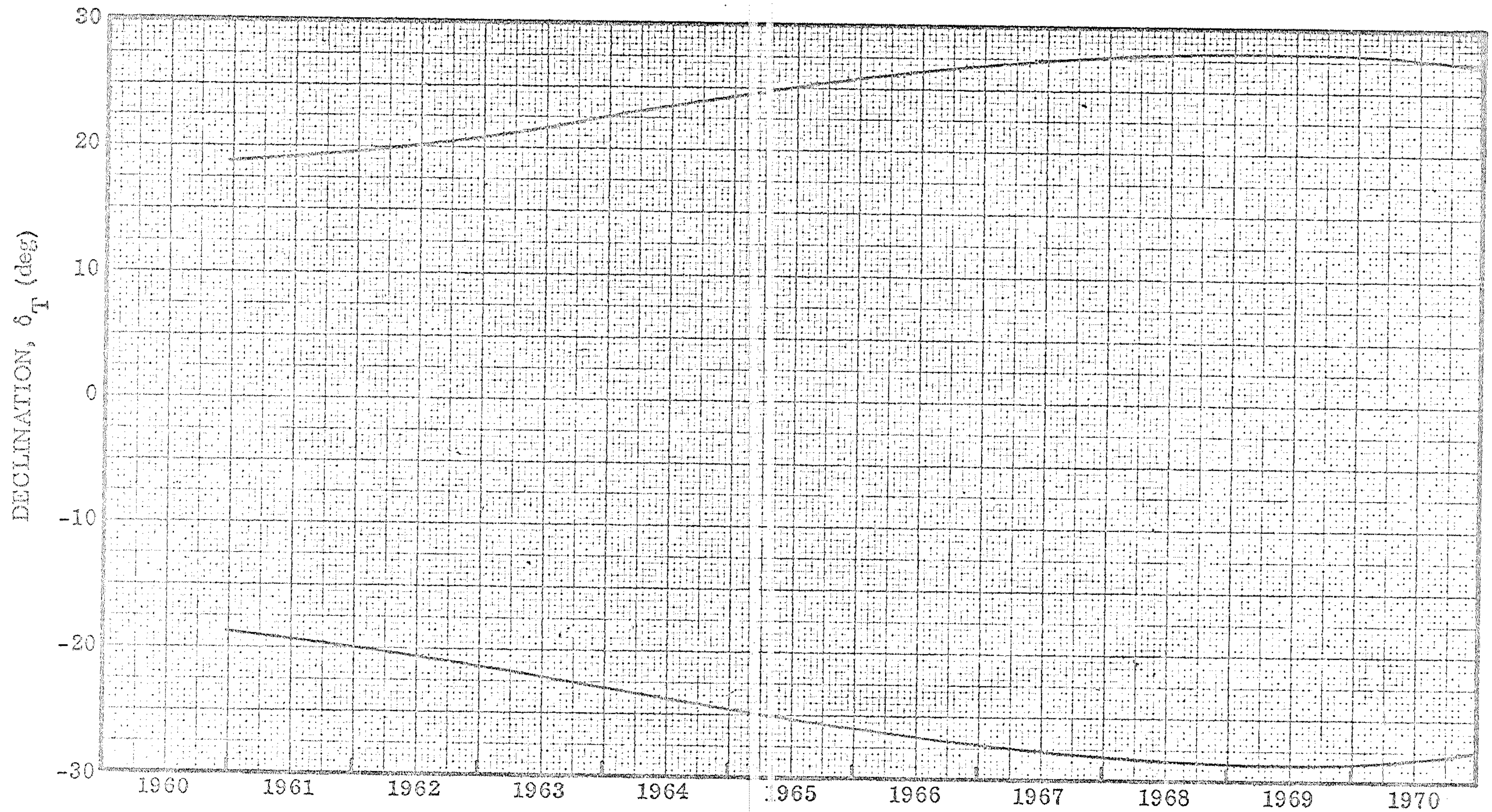


Figure 5-4. Maximum and Minimum Lunar Declination for the 1960 Decade

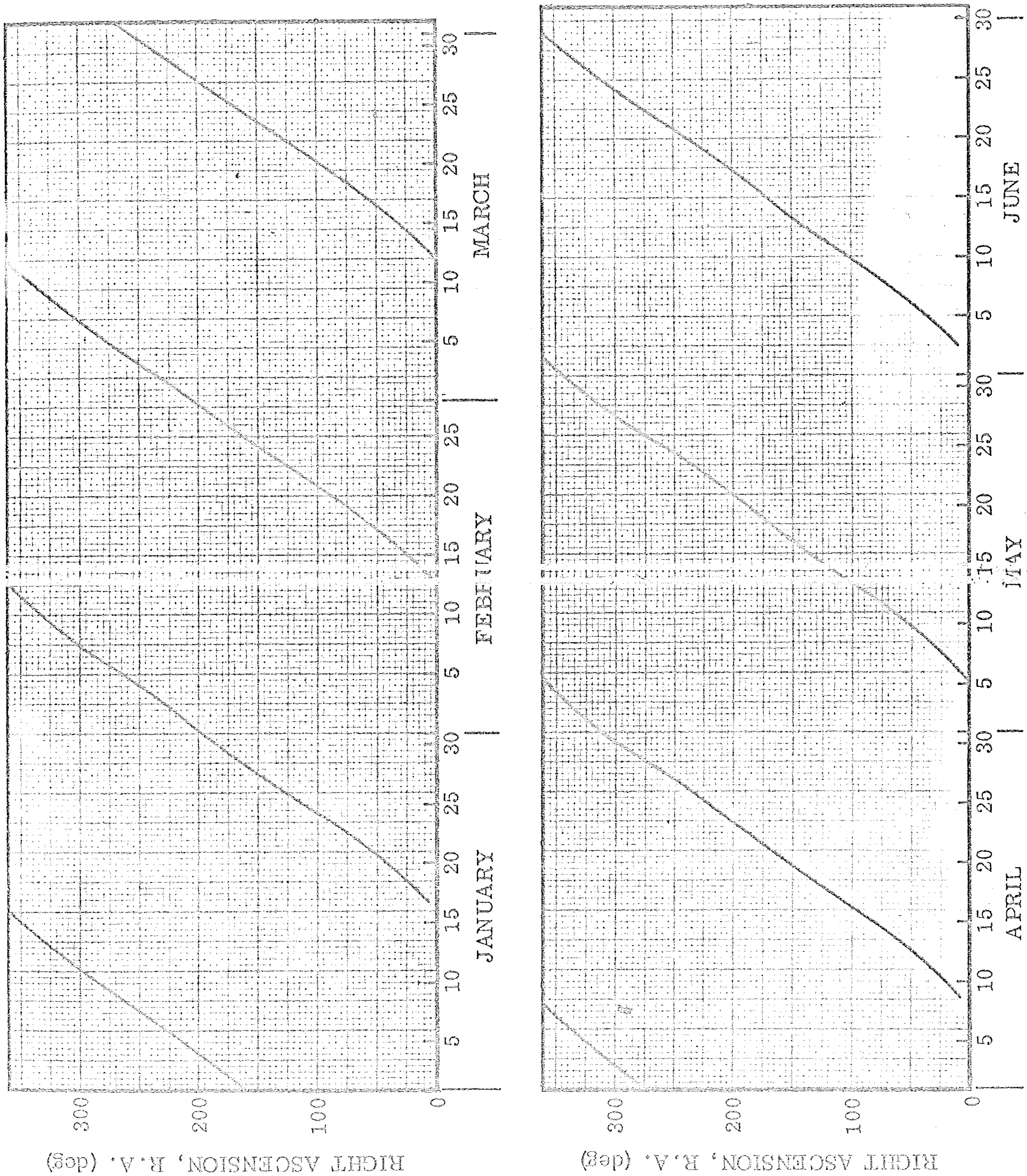


Figure 5-5. Lunar Right Ascension 1967 (Sheet 1 of 2)

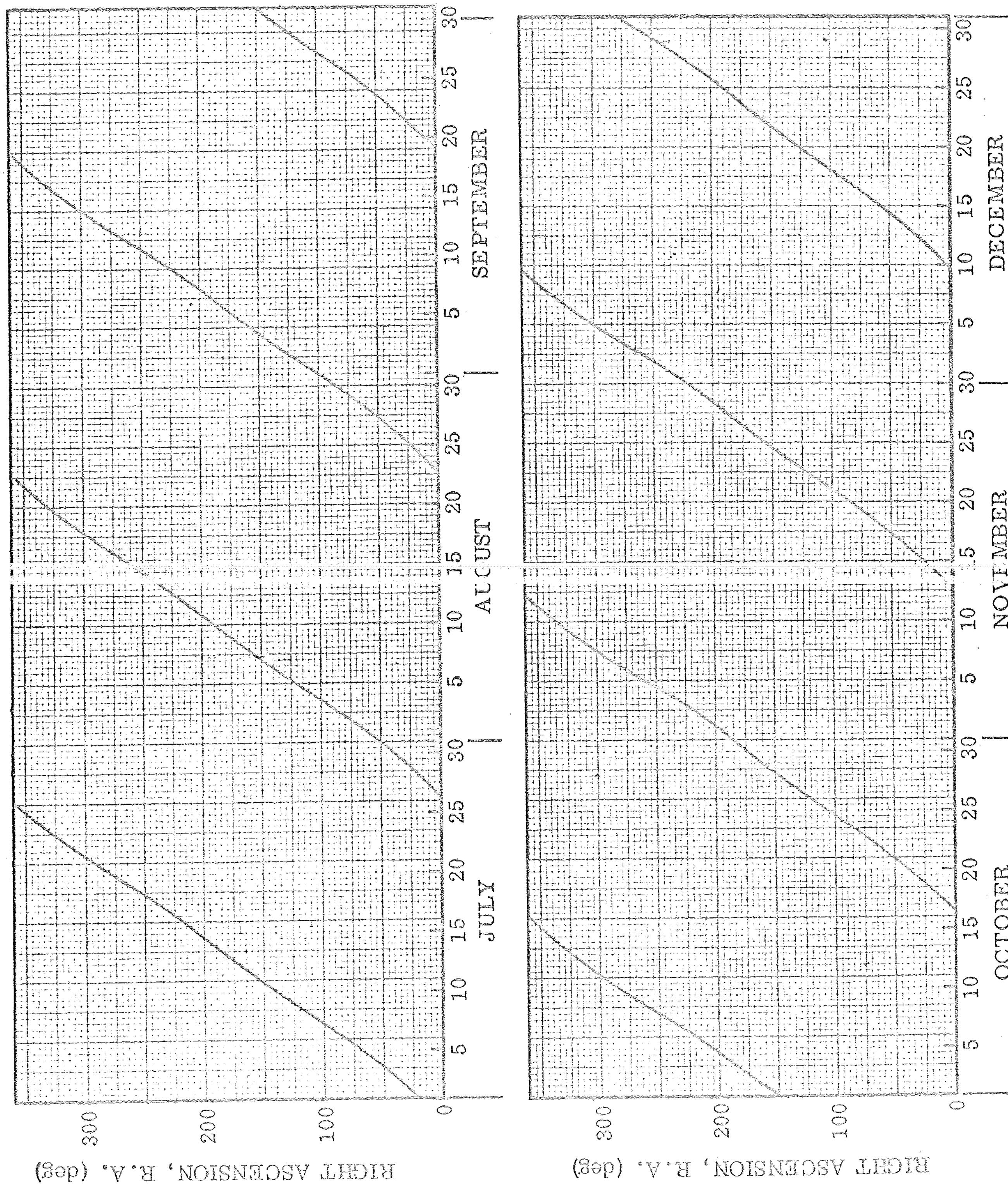


Figure 5-5. Lunar Right Ascension 1967 (Sheet 2 of 2)

5.2 PRE-INJECTION TRAJECTORY PHASE

The pre-injection trajectory phase can be described as a near-earth powered flight phase from booster launch to spacecraft injection. Parking orbit ascent in-plane geometry showing various angular relationships of the near-earth conic is illustrated in Figure 5-6. An expression relating the various angles indicated is presented in Equation 5-1.

$$\phi_{PO} = \phi_{LT} - \phi_{PT} - \phi_{B1} - \phi_{B2} + \eta_I \qquad (5\text{-}1)$$

During a given launch interval the position of the target vector is essentially invariant; however, the launch site moves in an easterly direction and reduces the launch-to-target angle, ϕ_{LT}. For lunar missions employing the parking orbit mode of ascent and having a 66-hour earth-moon transfer time, the injection true anomaly, η_I, is about 4 degrees. Prime control variable for balancing the in-plane angular relationships is parking orbit coast time, ϕ_{PO}. The first and second powered flight angles, ϕ_{B1} and ϕ_{B2} respectively, are mainly a function of boost vehicle characteristics and spacecraft weight for a given mission. Values of the parameters ϕ_{B1} and ϕ_{B2} are about 20 and 9 degrees respectively. The perigee-to-target angle, ϕ_{PT}, is about 170 degrees and can be determined from equation 5-2.

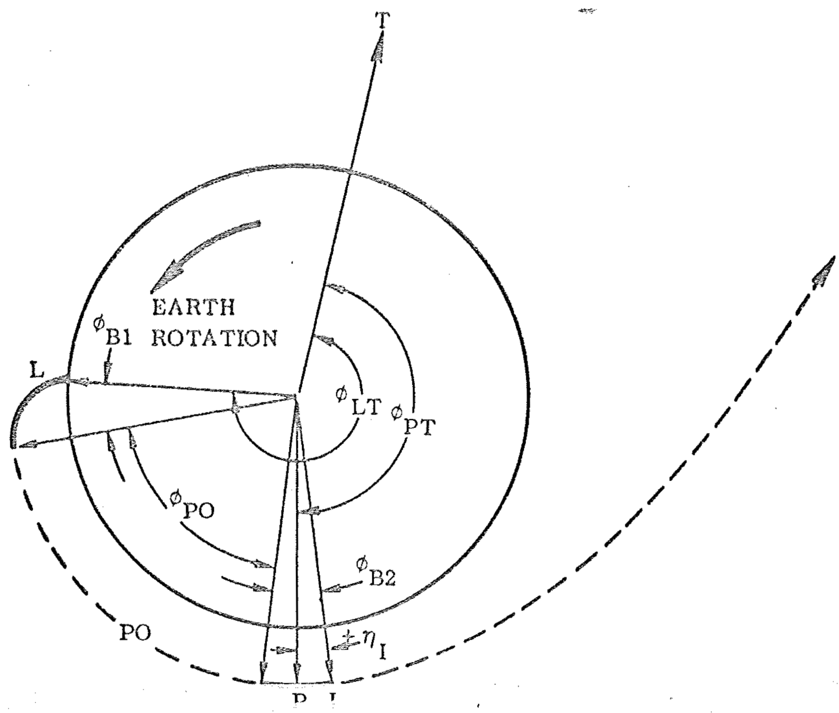
$$\phi_{PT} = \cos^{-1}\left[\frac{P - R_T}{eR_T}\right] \qquad (5\text{-}2)$$

$$P = R_P\,(1 + e)$$

$$e = 1 + \frac{R_P C_3}{\mu_E}$$

where

P	=	Semilatus rectum	C_3	=	Energy
R_P	=	Perigee distance	μ_E	=	Earth gravitational constant
e	=	Eccentricity	R_T	=	Earth-moon distance at encounter



ϕ_{B1} = First Powered Flight Angle ($\approx$ 20 deg)

ϕ_{PO} = Parking Orbit Angle (Variable)

ϕ_{B2} = Second Powered Flight Angle ($\approx$ 9 deg)

η_I = Injection True Anomaly ($\approx$ + 4 deg)

ϕ_{PT} = Perigee to Target Angle ($\approx$ 170 deg)

ϕ_{LT} = Launch to Target Angle

L = Launch Site

PO = Parking Orbit

P = Perigee

I = Injection

T = Target

Figure 5-6. Parking Orbit Ascent In-Plane Geometry

5.3 POST-INJECTION TRAJECTORY PHASE

The post-injection trajectory phase can be described as the earth-moon coast flight phase from spacecraft injection to the spacecraft terminal descent phase. Figure 5-7 is an illustration of the required earth-moon launch geometry for the post-injection trajectory phase.

Two position vectors that define the earth-moon coast flight plane orientation are the launch site vector declination, δ_L, and the target vector declination, δ_T. An expression that relates the launch-to-target in-plane angle, ϕ_{LT}, with the two position vectors and launch azimuth, Σ_L, is given by Equation 5-3.

$$\phi_{LT} = \sin^{-1}\left\{-\frac{C_2}{C_1} - \left[\left(\frac{C_2}{C_1}\right)^2 - \frac{C_3}{C_1}\right]^{1/2}\right\} \tag{5-3}$$

where

$$C_1 = 1 - \cos^2\delta_L \cdot \sin^2\Sigma_L$$

$$C_2 = -\cos\delta_L \cdot \sin\delta_T \cdot \cos\Sigma_L$$

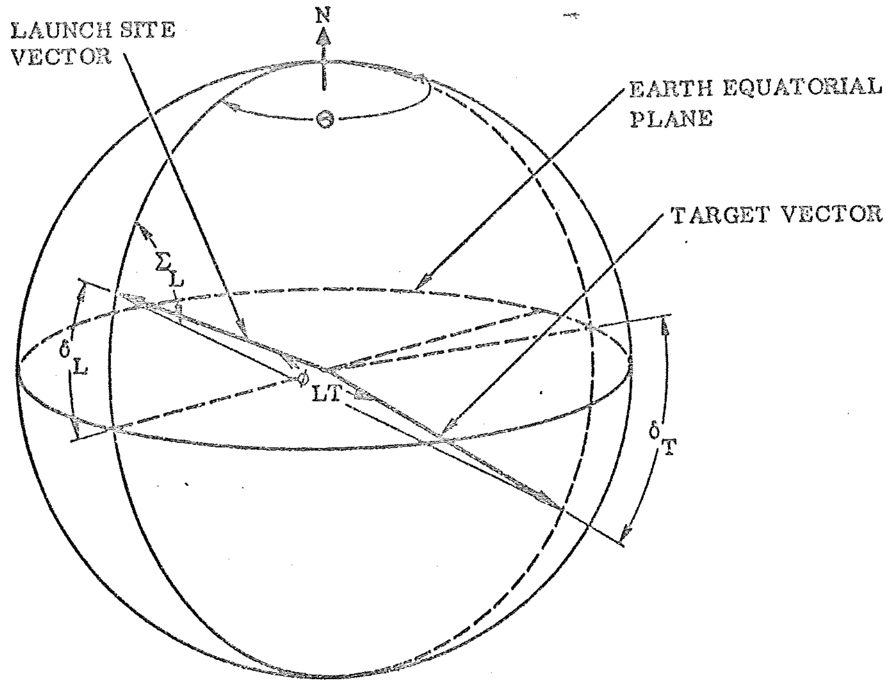
$$C_3 = \sin^2\delta_T - \sin^2\delta_L$$

Figure 5-8 presents a plot of ϕ_{LT} as a function of δ_T for a Σ_L range of 75 to 115 degrees and the Cape Kennedy launch site.

An expression that gives the launch-to-target hour angle, Θ, which is necessary for the determination of launch window length, is given by Equation 5-4.

$$\Theta = \sin^{-1}\left[\frac{\sin\phi_{LT}\sin\Sigma_L}{\cos\delta_T}\right] \tag{5-4}$$

Earlier an earth-moon flight time of 66 hours was mentioned. This transfer time is necessary to guarantee Goldstone tracking station visibility at lunar impact and required unbraked arrival velocity (due to retromotor capability). Earth-moon flight time, T_F, is a function of injection energy, C_3, injection perigee distance, R_P, and earth-moon distance at encounter, R_T.



Σ_L = Launch Azimuth

δ_L = Launch Site Vector Declination

ϕ_{LT} = Launch-to-Target In-Plane Angle

Θ = Launch-to-Target Hour Angle

δ_T = Target Vector Declination

Figure 5-7. Earth-Moon Launch Geometry

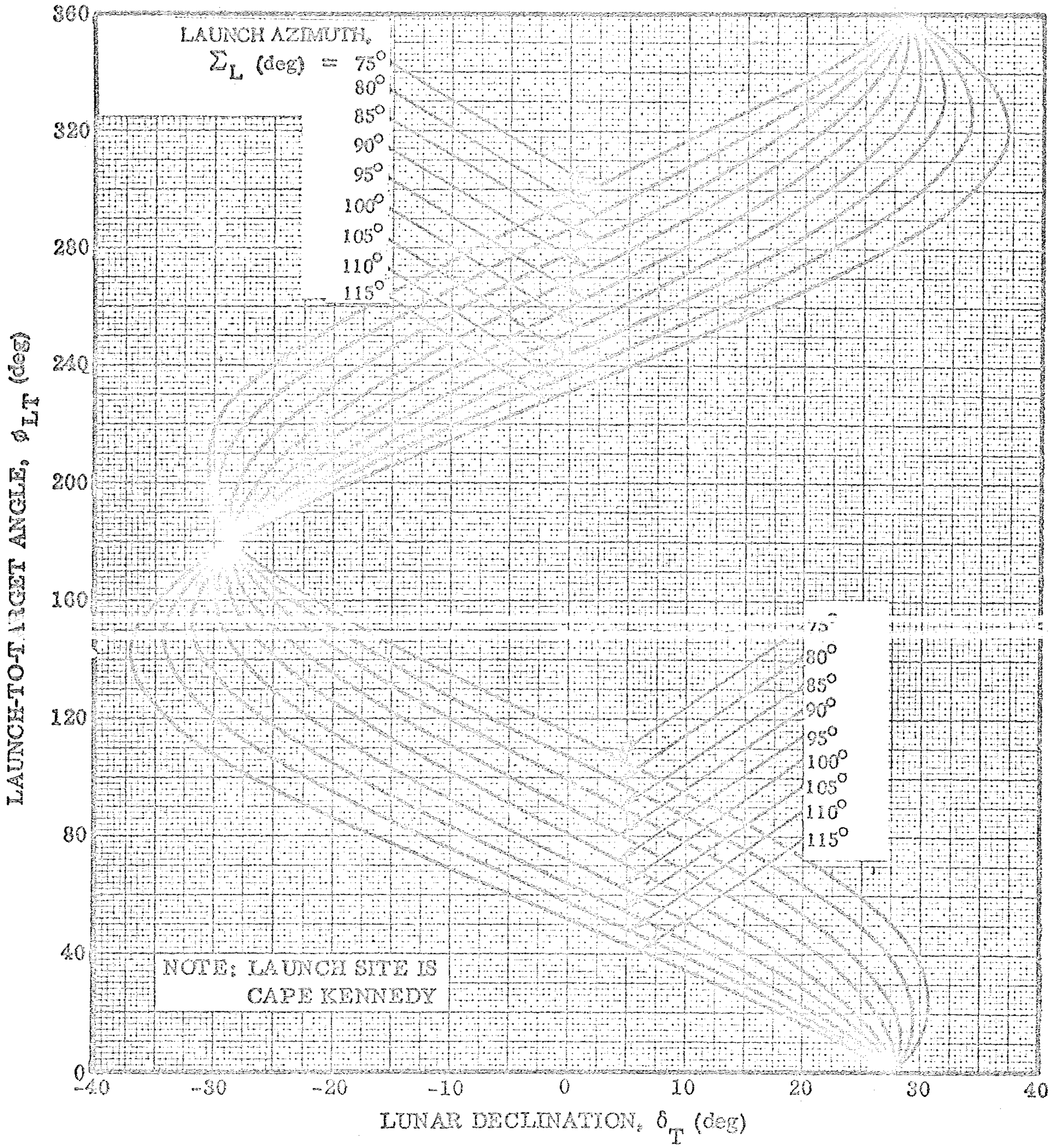


Figure 5-8. Launch-to-Target Angle Versus Lunar Declination

An expression that gives T_F as a function of the various parameters just mentioned (refer also to Equation 5-2) is presented in Equation 5-5.

$$T_F = \frac{a}{\sqrt{|C_3|}} \left(E - e \sin E \right) - \Delta t \qquad (5\text{-}5)$$

$$E = 2 \tan^{-1} \left[\sqrt{\frac{1-e}{1+e}} \tan \left(\frac{\phi_{PT}}{2} \right) \right]$$

$$a = \frac{\mu_E}{|C_3|}$$

where

a = Semimajor axis

E = Eccentric anomaly

Δt = Correction for lunar attraction

A plot of T_F as a function of C_3 for various R_T's is presented in Figure 5-6.

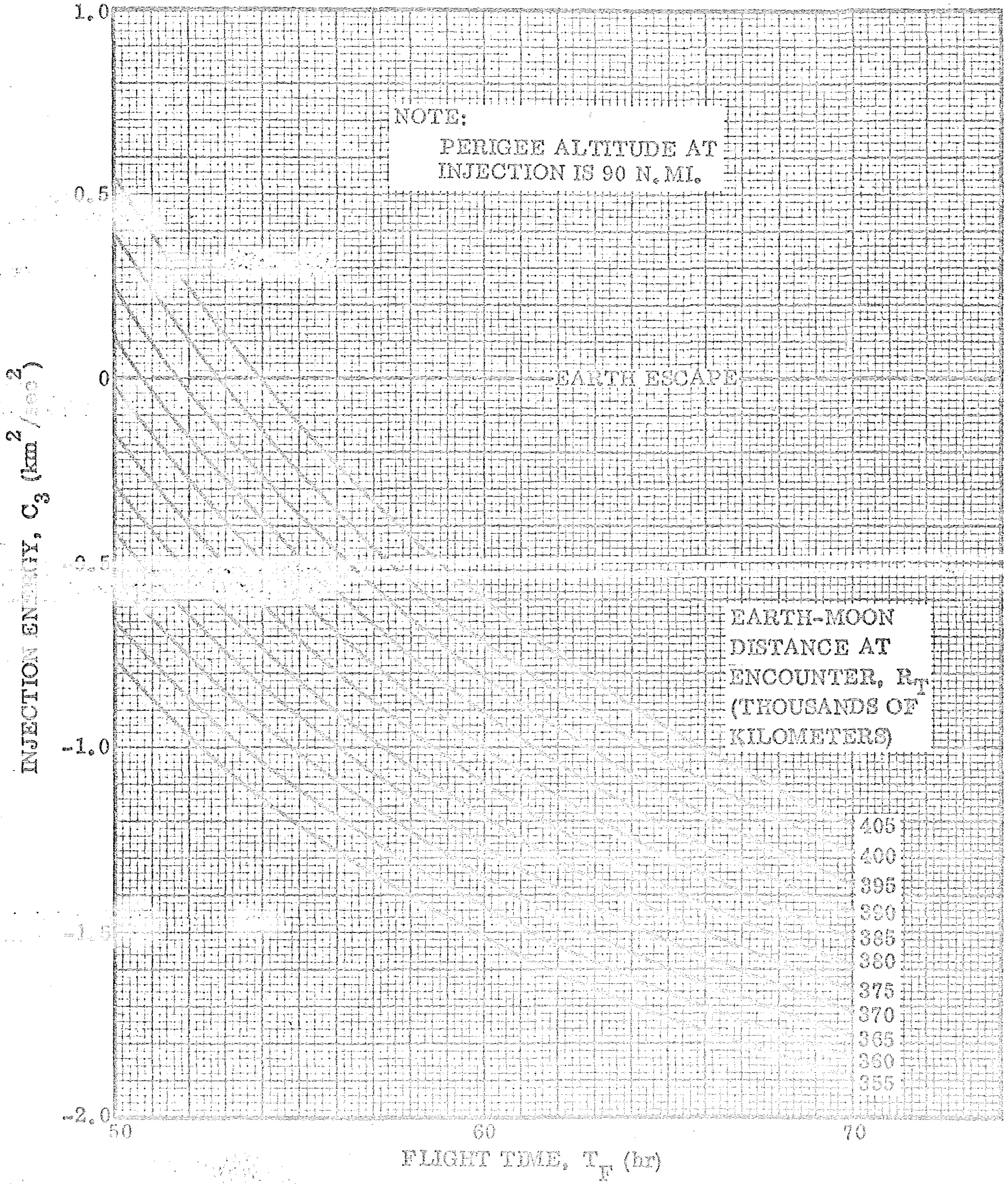


Figure 5-9. Earth-Moon Injection Energy Versus Flight Time

5.4 LAUNCH WINDOW

The time interval during a given day when it is possible to launch a vehicle is known as a launch window. Duration of the window is primarily a function of time-varying geometry and vehicle payload capability. A successive number of favorable launch window days is known as a launch period.

Since the earth-moon trajectory flight plane must contain the launch site and target vectors at launch time (see Figure 5-7) a time-varying launch azimuth, Σ_L, is required to account for the rotating launch site. Due to range safety considerations all parking orbit ascent flights are currently constrained to a launch azimuth sector between 78 and 115 degrees.

Launch window duration, t_{LW}, can be approximately determined from Equation 5-6 in conjunction with Equations 5-3 and 5-4 for a given lunar declination and specified launch azimuth limits. The launch window duration obtained may be further modified by other constraints such as payload capability, earth or moon shadow constraints, et cetera.

$$t_{LW} = \frac{\Theta_1 - \Theta_2}{\omega_E} \qquad (5\text{-}6)$$

where

Θ_1 = Hour angle at window opening

Θ_2 = Hour angle at window closing

ω_E = Earth rotation rate

SECTION 6

TRAJECTORY DESIGN CRITERIA

This section contains descriptions and data related to the rationale of powered-flight trajectory design. Primary emphasis is placed on identification of trajectory design constraints and on demonstration of an acceptable trajectory flight path and sequencing.

Fundamental to the trajectory design process is determination of vehicle subsystem performance and its effect on the trajectory. All elements of vehicle design must be compatible with the range of expected operating performance of each vehicle subsystem. Reference 12 serves to collect a standard set of subsystem performance dispersions that are used for both trajectory and guidance error studies.

Certain key parameters describing trajectory environment are presented and discussed in the following subsections. For a more comprehensive presentation of trajectory environment and documentation of design limits used in figures for this section, the reader is referred to Reference 13.

6.1 AERODYNAMIC LOADING

The trajectory parameter most often associated with aerodynamic loading is dynamic pressure (q), defined as:

$$q = \frac{1}{2}\rho \ V^2 \ (\text{lb/ft}^2) \qquad (6\text{-}1)$$

Dynamic pressure is primarily determined by the degree of trajectory lofting during the initial pitchover phase. Figure 6-1 presents time histories of dynamic pressure for both nominal and ± 3-sigma extreme flight conditions.

The product of dynamic pressure and angle of attach ($q\alpha$) is an indicator of bending moments imposed on the vehicle and is termed Aerodynamic Load Parameter (ALP).

The angle-of-attack time history is strongly influenced by both the pitch program and the applied wind. The pitch program has been designed with an angle-of-attack bias that will minimize ALP for anticipated launch winds and thus maximize launch availability.

6.2 AERODYNAMIC HEATING

A trajectory parameter that is indicative of the trajectory thermal environment is the product of dynamic pressure and velocity, termed heat flux parameter (HFP).

$$HFP = qV \tag{6-2}$$

The time integral of this parameter, which is indicative of total vehicle heating, is termed heat parameter (HP).

$$HP = \int_{t=0}^{t} qV\,dt \tag{6-3}$$

This simplification is generally valid for turbulent heating and is used as a rule of thumb in trajectory design. Final trajectory heating verification is based on detailed thermodynamic analysis of the vehicle in the appropriate flight environment.

Like dynamic pressure, heat parameter is primarily determined by the degree of trajectory lofting during the initial pitchover phase.

Figures 6-2 and 6-3 present time histories of HFP and HP respectively for nominal and design flight conditions, minimun and maximum. From Figure 6-3, the maximum value of HP is 0.945×10^8 lb/ft. The nominal limiting value of HP is approximately 0.94×10^8 lb/ft. However, this trajectory design constraint is an indicator only, used in the early Surveyor trajectory design. A detailed thermodynamic analysis of the vehicle has shown that aerodynamic heating is within allowable limits.

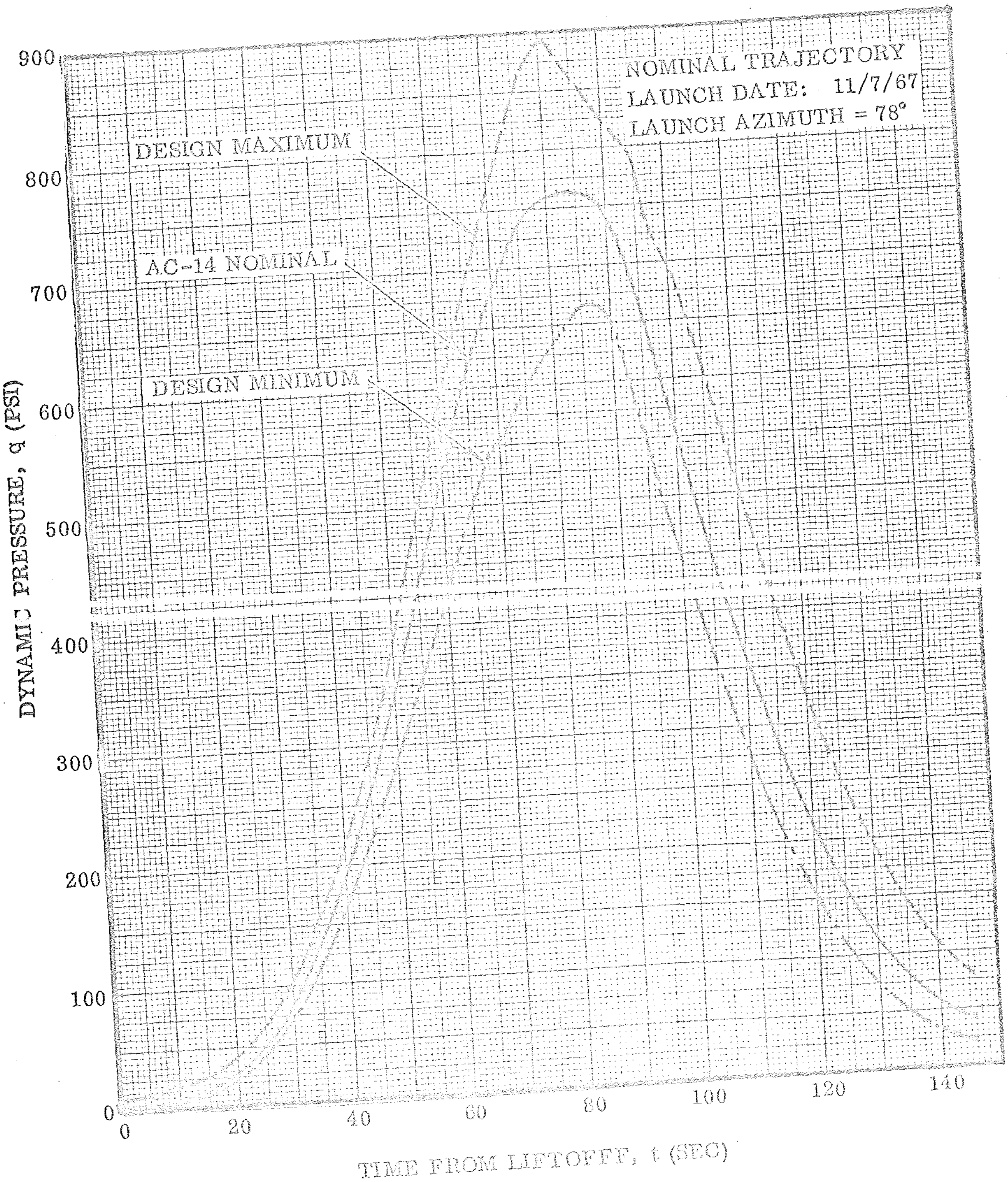


Figure 6-1. Dynamic Pressure Versus Flight Time

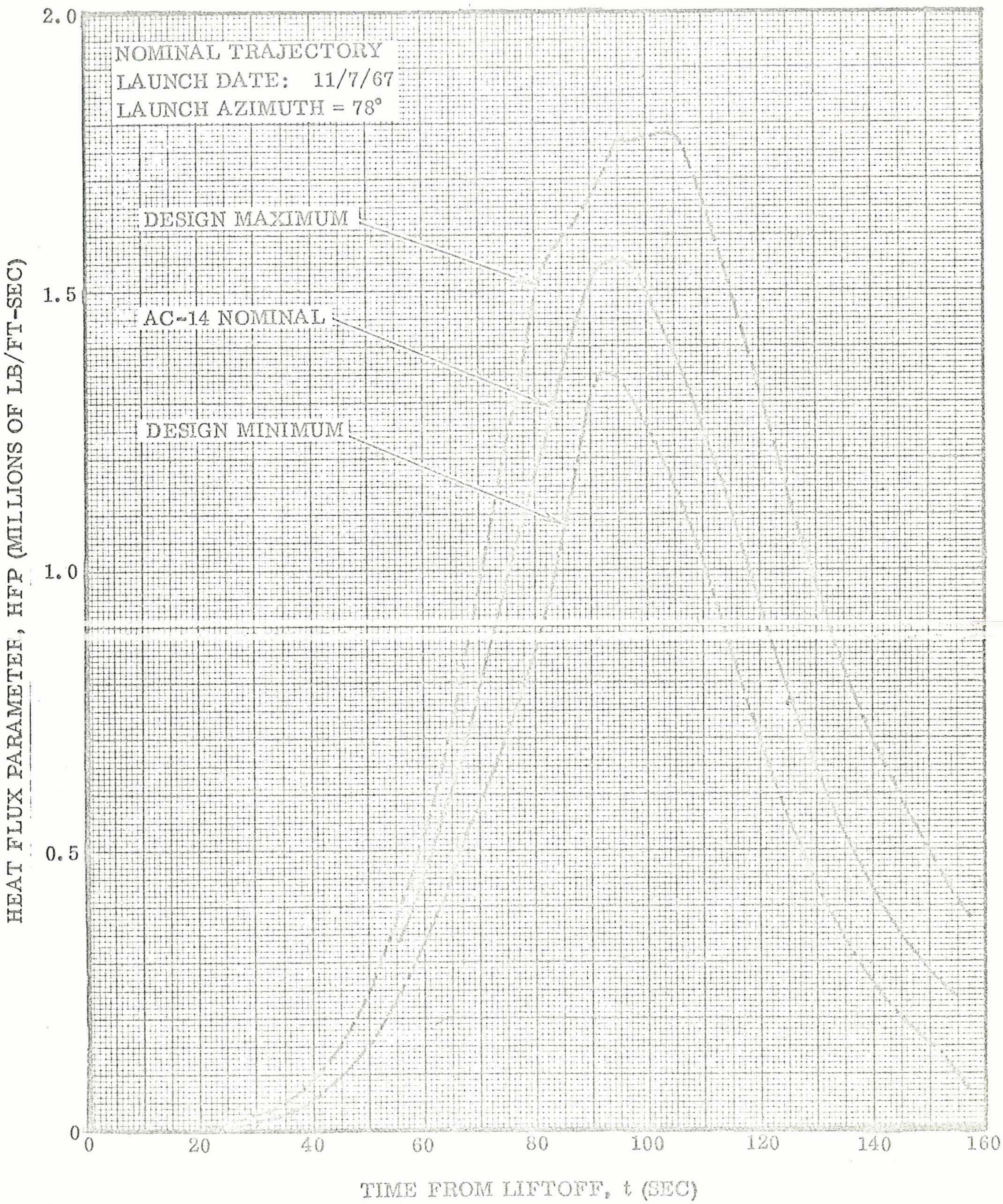


Figure 6-2. Heat Flux Parameter Versus Flight Time

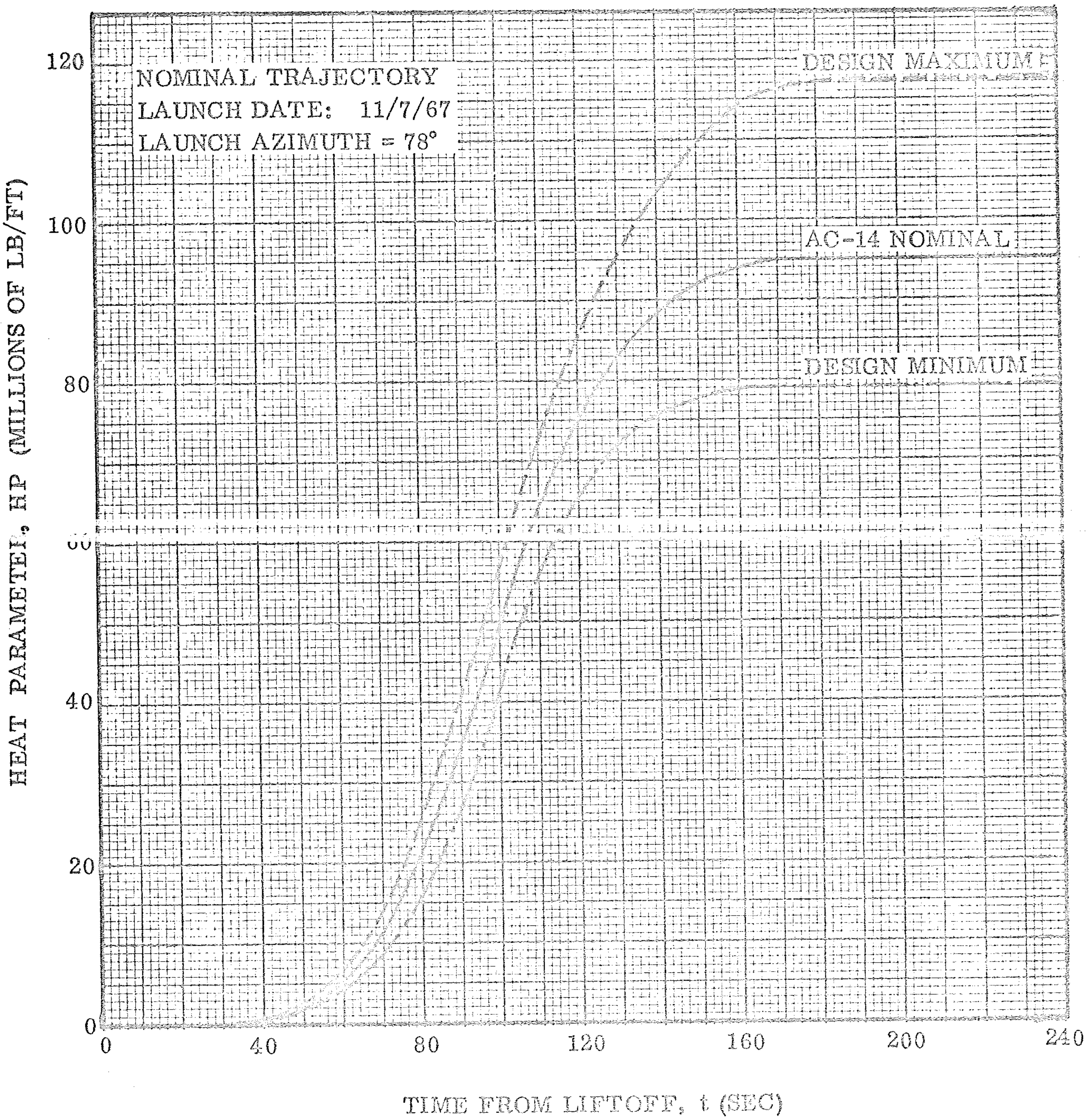


Figure 6-3. Heating Parameter Versus Flight Time

6.3 STAGING CRITERIA

6.3.1 BOOSTER ENGINE STAGING

All Atlas propulsion units are operative at time of liftoff. On-board accelerometers sense acceleration. When a nominal value of 5.70 g's is reached, a guidance discrete is issued cutting off the two booster thrust chambers (BECO). Following a 3.1-second delay for booster thrust decay, the booster propulsion unit is jettisoned.

Table 6-1 illustrates the accelerometer settings. The nature of the guidance computer is such that the issuance of the primary discrete will occur following a complete compute cycle. The change in acceleration during each compute cycle is approximately 0.15 g at the time of staging. The dispersion about the 5.7 g cutoff setting is reduced to one half of a compute cycle (±0.08 g) by biasing the guidance computer cutoff setting.

The staging level for the backup system was chosen so that, nominally, it will not preempt the primary system. The backup accelerometer is set at 5.90 g's and the 3-sigma deviation is ±0.20 g.

Table 6-1. Booster Staging Accelerometer Settings

	5.62 g's	5.70 g's	5.78 g's	
Guidance Accelerometer (Primary)	-3 Sigma	Nominal	+3 Sigma	
		5.70 g's	5.90 g's	6.10 g's
Backup Accelerometer		-3 Sigma	Nominal	+3 Sigma

The selection of booster staging acceleration level for Atlas/Centaur was based on the following criteria:

a. Maximize payload capability.

b. Atlas load < design allowable.

c. Net positive suction head (NPSH) > design minimum.

The staging level of 5.7g's was selected in the early phase of the Centaur development effort. At that time, booster pitch program design was predicated on the requirement for maximum payload capability and was not substantially constrained by aerodynamic heat limits. Under those conditions, the optimum staging level was near 5.7 g's.

The second factor that must be considered in establishing booster staging level is the Atlas load limit. The allowable total gross weight above Station 519 at BECO cannot exceed 46,939 pounds (Reference 4). The following assumptions were used in calculating the allowable weight:

Axial acceleration	5.7 ± 0.08 g
Tank skin temperature	171° F @ Sta. 789.95
Axial drag force	1330 lb
Applied bending moment	0.761×10^6 in.-lb
LO_2 tank pressure	32.0 psig

From Reference 4, the nominal weight above Station 519 is 41,010 pounds.

The third and final factor affecting selection of BECO staging acceleration is sustainer LO_2 pump NPSH. Briefly, it is required that NPSH be above 13 feet. Section 8.1.1 presents a more detailed discussion of sustainer NPSH analysis.

6.3.2 SUSTAINER ENGINE STAGING

The criterion for sustainer engine staging is propellant depletion, either oxidizer or fuel. SECO command is initiated (nominally) by a fuel manifold pressure switch set to activate at approximately 650 psia. This mode of sustainer staging results in maximum net vehicle payload capability.

Section 8.1.2 presents SECO propulsion characteristics in some detail.

6.3.3 INSULATION PANEL STAGING

Centaur propellants are protected from aerodynamic heating during the ascent trajectory by insulation panels. These panels, which shroud the hydrogen tank cylindrical walls, are jettisoned by an Atlas programmer discrete at BECO + 45 seconds.

Three criteria are considered in establishing insulation panel jettison time, namely:

a. Maximum vehicle payload capability
b. Thermal load constraint
c. Aerodynamic load constraint

The first factor involves a tradeoff between payload gain due to reduced propellant vaporization and payload loss due to lower vehicle acceleration as insulatation panel jettison time increases. Figure 6-4 provides a typical example of this tradeoff.

After insulation panel jettison, heating to the LH_2 tank wall increases abruptly due to sudden exposure to aerodynamic and solar heating and to sublimation of ambient air. Vapor bubbles are generated in the liquid, causing the propellant volume to swell and the liquid level to rise in the tank. Panel jettison is timed so that the liquid level does not overflow into the boiloff gas vent at the forward end of the tank. This requirement is satisfied when heat flux parameter is less than 65,000 lb/ft-sec (3-sigma). However this factor was not considered in the final selection of panel jettison time. (See Figure 6-5.

The final, and limiting, constraint on panel jettison is aerodynamic loading. It is a requirement that the panels be jettisoned sufficiently late in flight so that aerodynamic loads do not inhibit separation. Studies have shown this requirement is satisfied when the product of dynamic pressure and angle of attack is equal to or less than 20 psf-deg (3 sigma). Figure 6-6 shows that this aerodynamic load factor is satisfied with a jettison time of BECO + 45 seconds.

6.3.4 NOSE FAIRING STAGING

Staging of the Centaur nose fairing (NF) is accomplished by an Atlas programmer discrete at a preset time after BECO of 75 seconds. The time at which the NF may be jettisoned is determined by the ability of the payload to withstand the aerodynamic heating it will receive after being exposed to the flight environment. Aerodynamic heating, in the altitude range where NF jettison is feasible, is directly proportional to heat flux parameter (HFP). A 3 sigma maximum value of HFP at NF jettison has been established by Hughes Aircraft Company for the Surveyor spacecraft. This value is 295 lb/ft-sec. Figure 6-7 presents the variation of HFP with time after BECO and shows the earliest allowable NF jettison time to be BECO + 70 seconds. Thus the present staging time of BECO + 75 seconds allows a 5-second contingency for configuration changes.

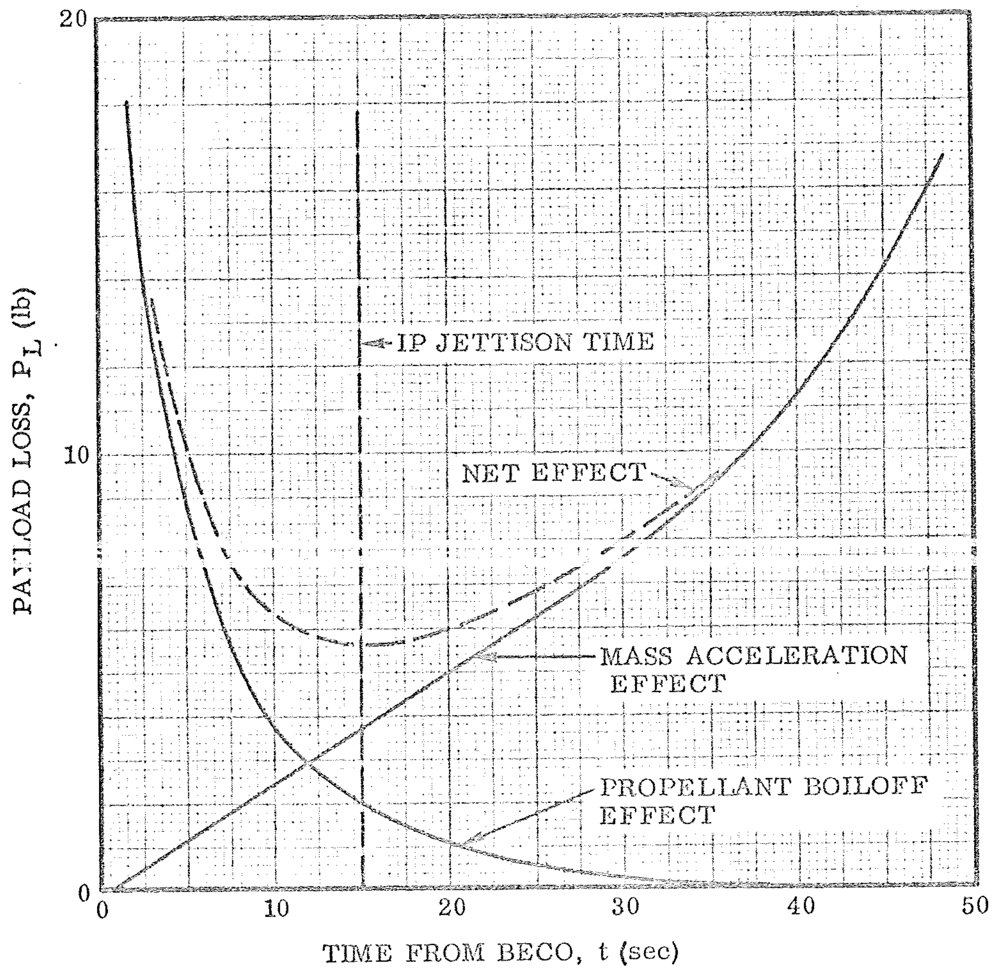


Figure 6-4. Typical Effect on Payload Capability of Insulation Panel Jettison Time

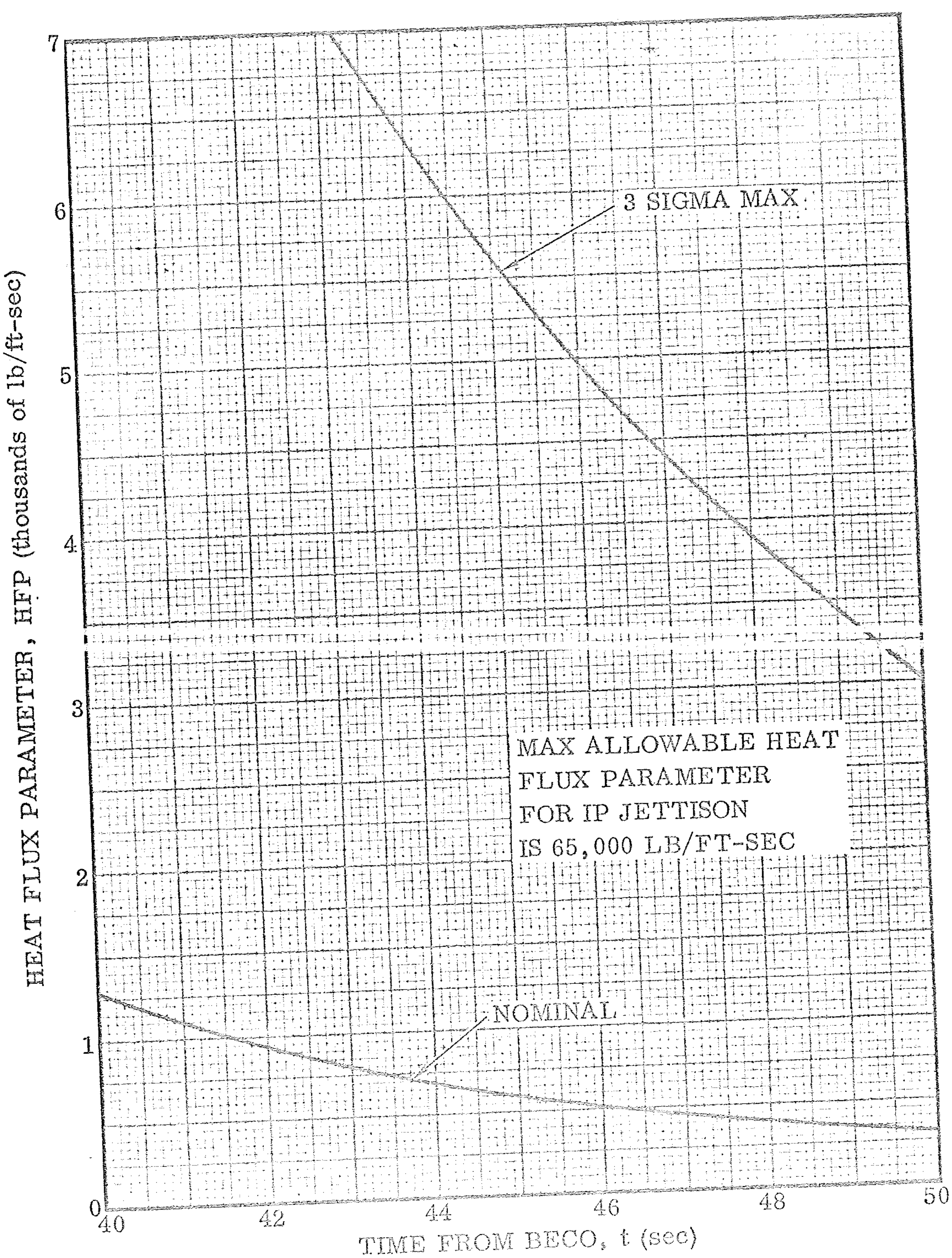


Figure 6-5. Effect of Insulation Panel Jettison Time on Heat Flux Parameter for Pitch Program 211

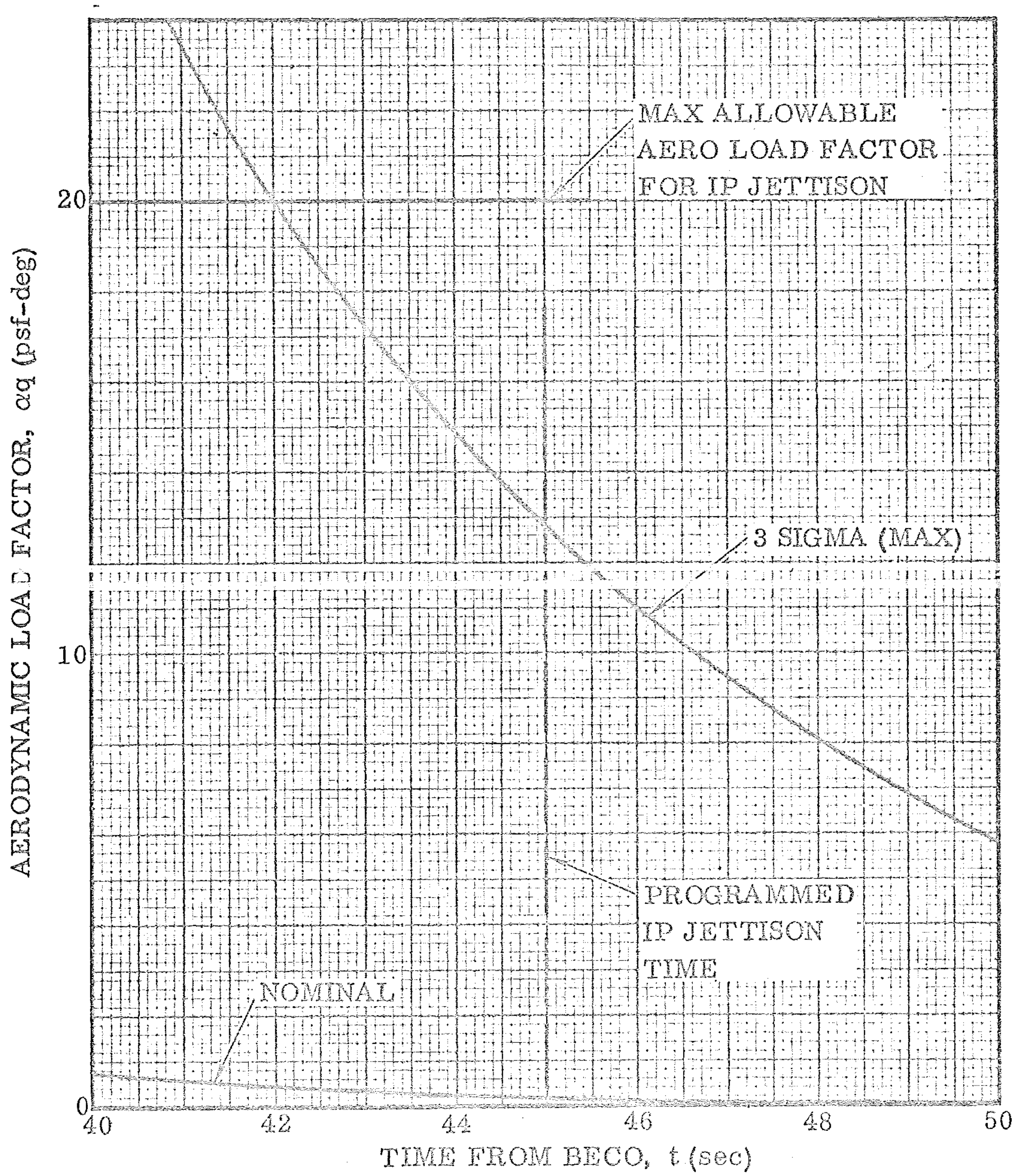


Figure 6-6. Aerodynamic Environment Insulation Panel Staging Criterion Based on Pitch Program 211

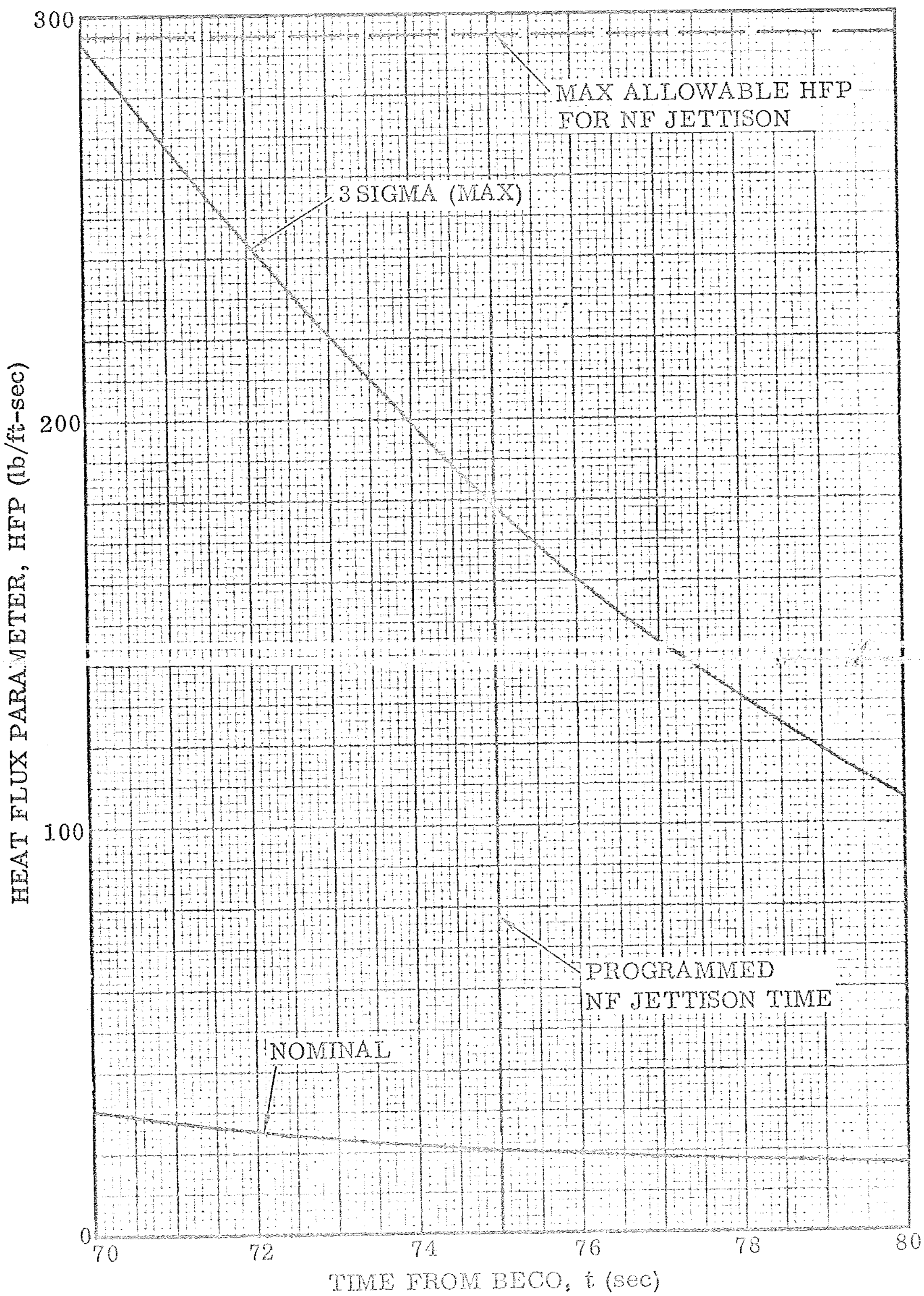


Figure 6-7. Thermal Environment Nose Fairing Staging Criterion Based on Pitch Program 211

6.4 PITCH PROGRAM DESIGN

6.4.1 BOOSTER PITCH PROGRAM

Design of the booster pitch program would ideally involve an optimization between flight path selection and structural weight required to sustain the flight path environment. For Centaur, this process has been bypassed since Atlas was an off-the-shelf vehicle. Thus booster pitch program design was based on predefined vehicle structural capability.

Aerodynamic loads are minimized during booster phase by flying a near-zero angle-of-attack (α) boost trajectory. The pitch program is actually biased to statistically minimize α in the presence of anticipated real wind conditions.

Superimposed on the near-zero α requirement is a nominal aerodynamic heat parameter limit used in the initial trajectory design. Figure 6-8 illustrates the sensitivity of payload capability to HP.

6.4.2 SUSTAINER AND CENTAUR PHASE PITCH PROGRAMS

The sustainer- and Centaur-phase pitch programs are shaped to maximize payload capability for prescribed end conditions. These pitch functions are generated in flight by the guidance computer based on instantaneous present position and target conditions.

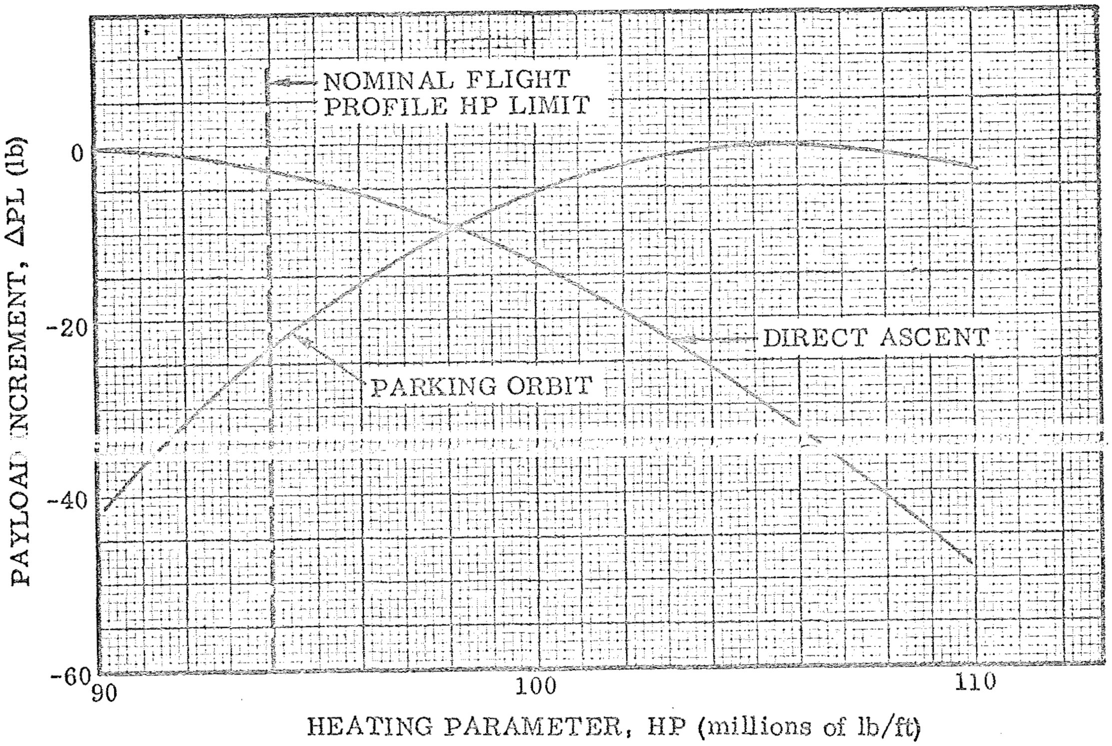


Figure 6-8. Effect of Pitch Program on Payload Capability

SECTION 7

VEHICLE CONFIGURATION

The launch vehicle is a two-stage configuration consisting of an Atlas SLV-3C first stage and a Centaur second stage. The Atlas by itself is referred to as a 1-1/2 stage configuration, a term used to describe a vehicle consisting of one set of propellant tanks with some propulsion units jettisonable during flight.

7.1 FIRST STAGE DESCRIPTION

The Atlas SLV-3C vehicle is a modification of the LV-3C vehicle previously used for Atlas/Centaur flights. The modification consists of a 51-inch extension of the propellant tank and relocation of the intermediate bulkhead to maintain the proper oxidizer/fuel ratio. Other minor changes were made to external equipment fairings. To accommodate the additional propellant weight the NAA/Rocketdyne MA-5 propulsion system was modified to provide an additional 6000 pounds of booster thrust and a sustainer engine thrust increment of 1000 pounds. Each of the booster engines now provide 168,000 pounds nominal thrust (sea level) and the sustainer engine 58,000 pounds. The vernier thrust chambers still provide 670 pounds each.

The interstage adapter, mating the Centaur to the Atlas, is attached permanently to the Atlas stage. The Centaur vehicle is released from the first stage by severing the interstage adapter by use of a flexible linear shaped charge attached around the inner perimeter of the adapter just below the Centaur tank interstage adapter attachment point. Eight retrorockets located at the extreme aft end of the cylindrical portion of the Atlas fuel tank provide the force for separating the Atlas section from Centaur.

The controlling document for the Atlas vehicle is Reference 14.

The general arrangement of Atlas is presented on Figure 7-1.

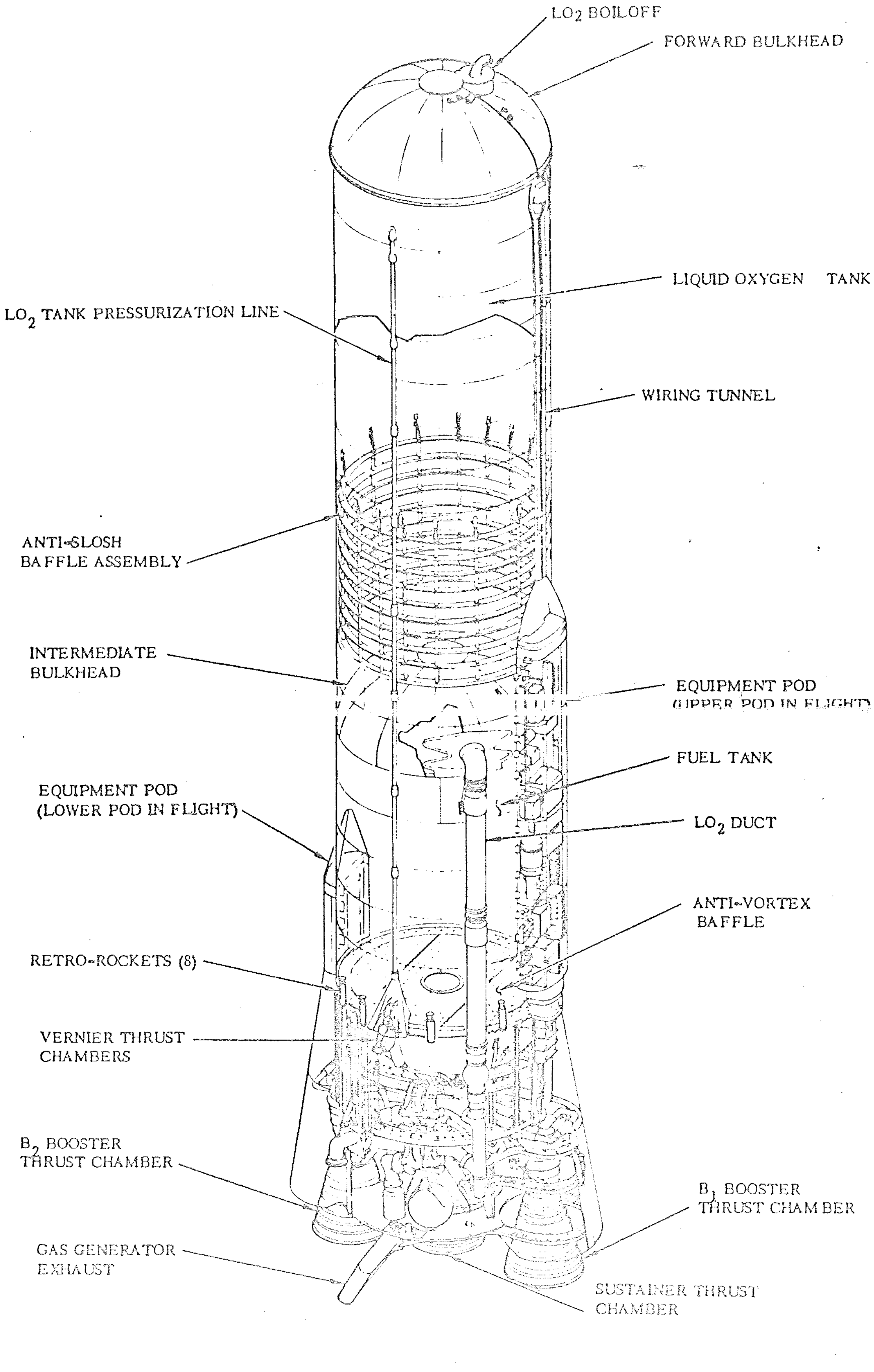


Figure 7-1. General Arrangement of Atlas

7.2 SECOND STAGE DESCRIPTION

The Centaur vehicle is a high-specific-impulse space vehicle powered by two Pratt & Whitney RL10A main engines rated at 15,000 pounds thrust each (with a restart capability). These engines burn liquid oxygen and liquid hydrogen and are located on the aft bulkhead of the liquid oxygen tank.

The Centaur tank skins are made of type 301 stainless steel, which is the same material used for the Atlas tanks. The propellant tanks are pressurized and form the basic shape of the Centaur. The payload, guidance, and electronic equipment packages are mounted on the forward bulkhead of the liquid hydrogen tank.

External thermal insulation panels are provided for the liquid hydrogen tank to minimize the effects of aerodynamic heating and to reduce the liquid hydrogen boiloff to an acceptable level. The panels are jettisoned after ascent through the atmosphere.

The nose fairing provides thermal and aerodynamic loading protection to the equipment located within the spacecraft compartment at the forward end of the vehicle. The nose fairing is jettisoned following jettison of the insulation panels.

The controlling document for the configuration of the Centaur vehicle is Reference 15.

The general arrangement of Centaur is presented on Figure 7-2.

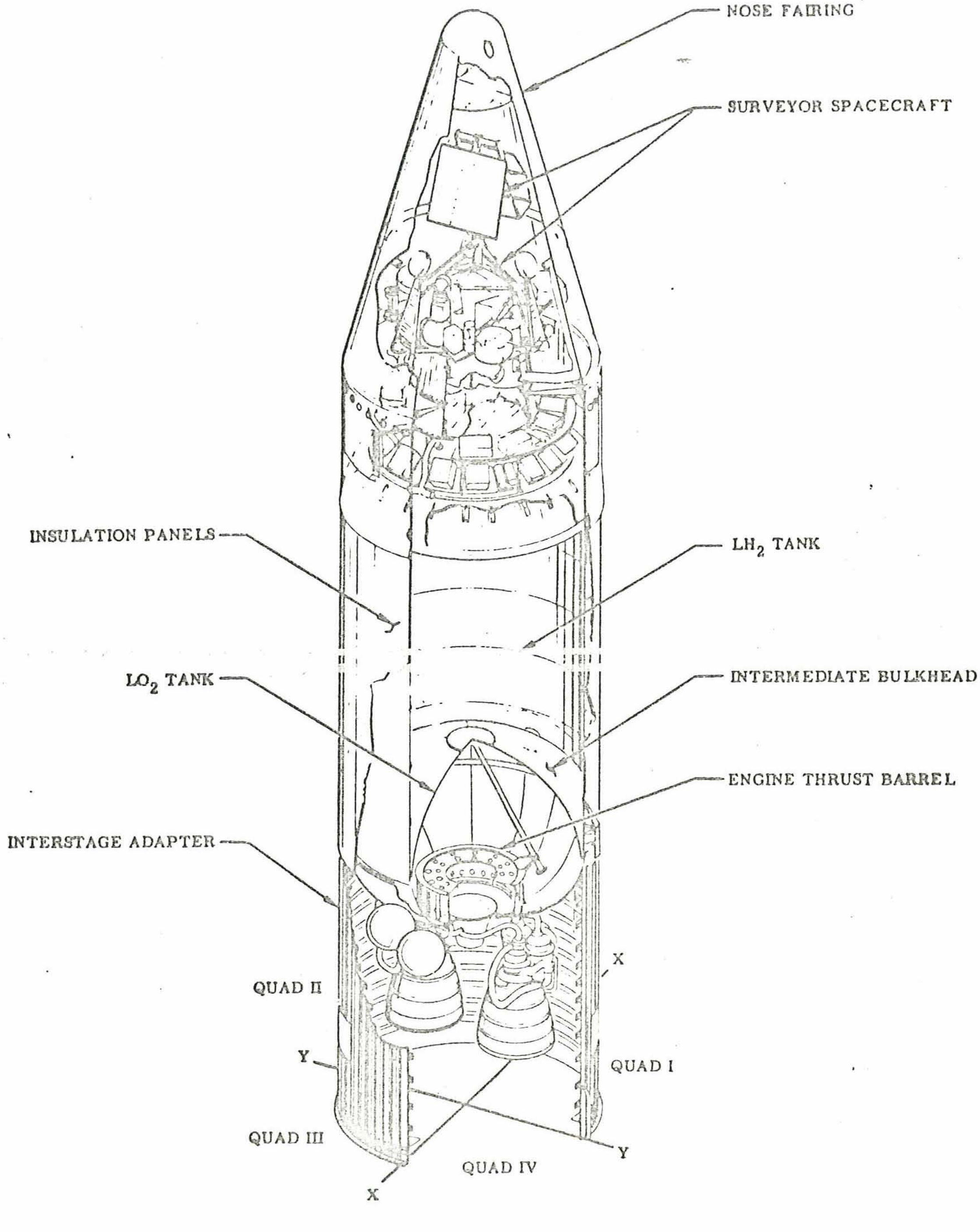


Figure 7-2. General Arrangement of Centaur

7.3 SPACECRAFT DESCRIPTION

The SC-6 spacecraft (A-21 SERIES) is a thin-walled tubular spaceframe housing various electronic and mechanical equipment. Figures 7-3 and 7-4 present the general arrangement of the spacecraft.

Spacecraft equipment can be broken down into the following systems: 2) structure, b) propulsion, c) flight control, d) electrical power, e) telecommunications, f) command decoding, g) engineering instrumentation and signal processing, and h) television.

The payload is composed of an alpha scattering experiment; magnets; an auxiliary battery and control unit; survey television camera; a number of temperature sensors, accelerometers, strain gauges, and current sensors; and an engineering signal processor.

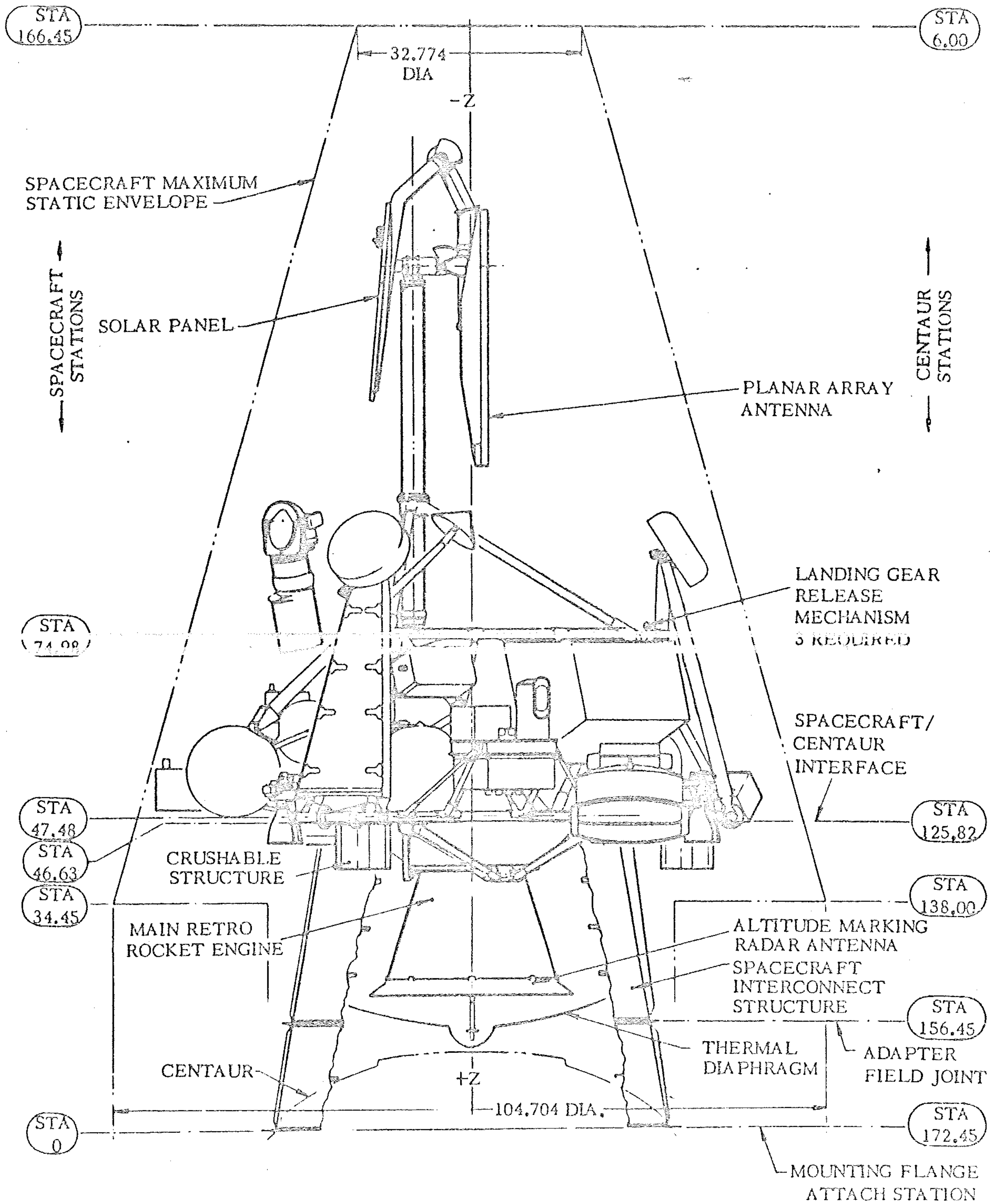


Figure 7-3. General Arrangement of SC-6 Spacecraft, Launch Configuration

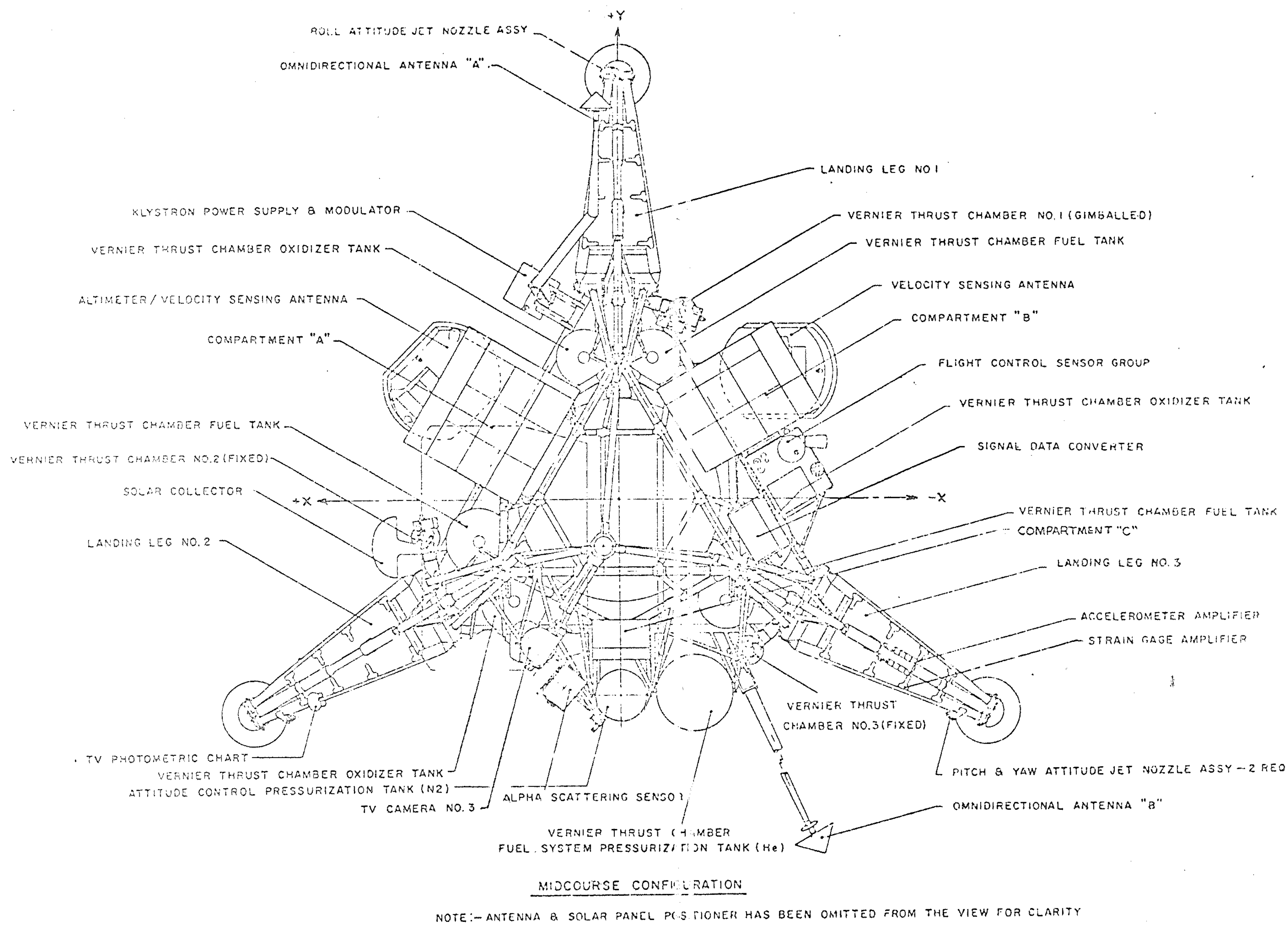


Figure 7-4. General Arrangement of SC-6 Spacecraft, Midcourse Configuration

SECTION 8

PROPULSION

The following paragraphs describe the Atlas and Centaur propulsion models and nominal operating characteristics.

8.1 ATLAS PROPULSION MODEL

The Atlas stage propulsion system consists of a NAA/Rocketdyne MA-5 propulsion system with two booster thrust chambers, a sustainer thrust chamber, and two vernier thrust chambers. The engines burn a propellant combination of liquid oxygen and RP-1 fuel. All propulsion units are operative at time of liftoff. The sustainer and vernier engines continue firing following shutdown and jettisoning of the two booster engines. At propellant depletion the sustainer and vernier units are shut down and jettisoned with the Atlas propellant tanks.

Nominal Atlas propulsion characteristics are summarized in Table 8-1. The data were obtained from Reference 16. This reference also contains the propulsion parameters used in detailed propulsion simulations for precision integrated trajectories.

Figures 8-1 and 8-2 present Alas thrust and specific impulse as a function of altitude. The data reflect nominal inflight engine operation as predicted via a detailed trajectory computation and propulsion system performance simulation.

Table 8-1. Summary of Atlas Propulsion Characteristics

PROPULSION PARAMETER	NOMINAL[1]	3σ DISPERSION
Booster S. L. Thrust	336,000 lb	±2945 lb
Booster S. L. Specific Impulse	253.83 sec	±2.33 sec
Booster S. L. Mixture (O_2/RP-1)	2.29	±0.038
Sustainer S. L. Thrust	58,000 lb	±910 lb
Sustainer S. L. Specific Impulse	215.39 sec	±2.72 sec
Sustainer S. L. Mixture Ratio (O_2/RP-1)	2.27	
Vernier S. L. Thrust[2]	1352.8 lb	2 %
Vernier S. L. Specific Impulse	191.4 sec	6.47 sec
Vernier S. L. Mixture Ratio (O_2/RP-1)	1.676	
Booster Lube Oil Consumption	0.95 lb/sec	
Sustainer Lube Oil Consumption	0.150 lb/sec	

[1] Engine log data.
[2] Vernier engines are canted 41 degrees outboard.

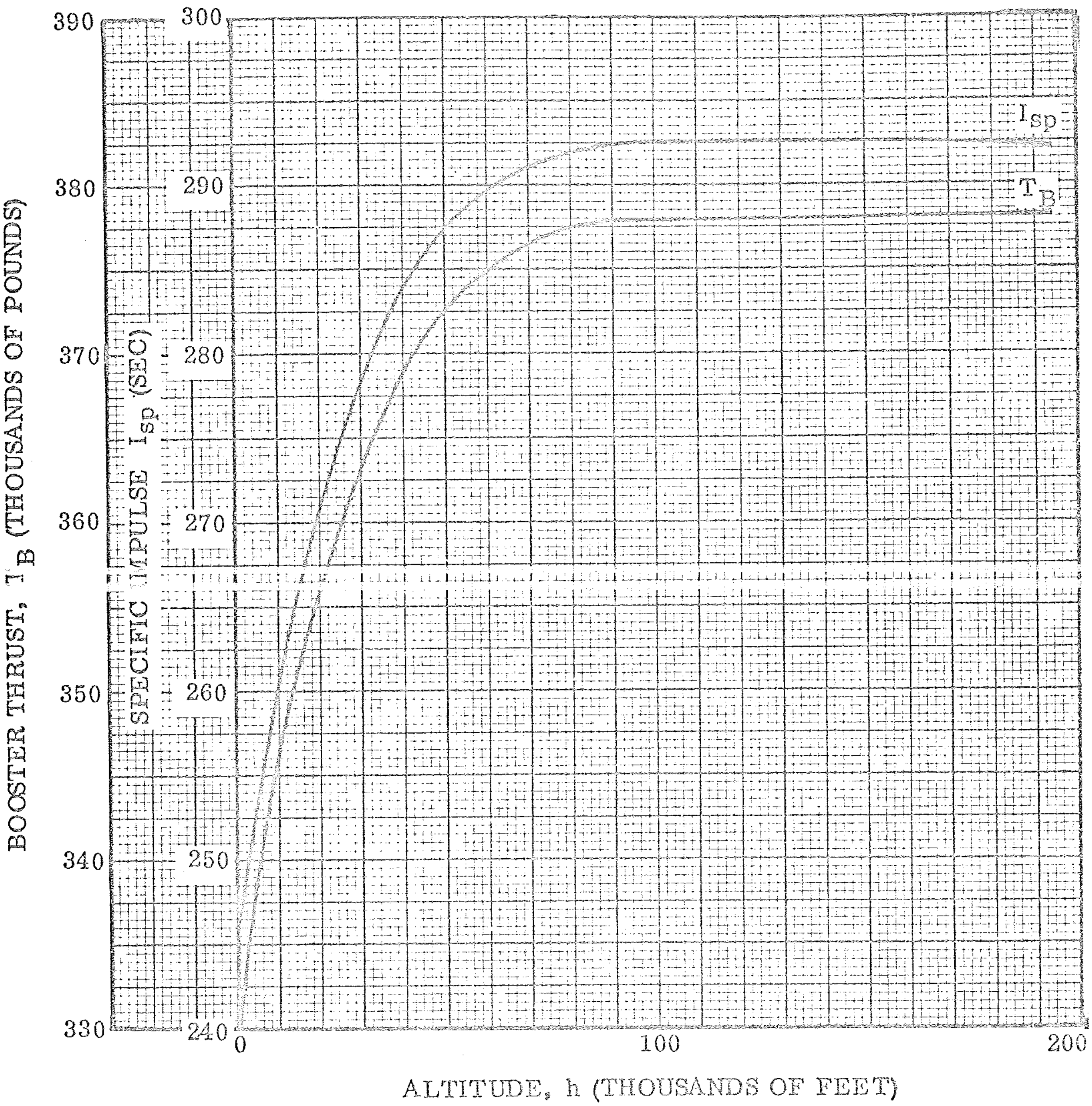


Figure 8-1. Atlas Booster Thrust and Specific Impulse

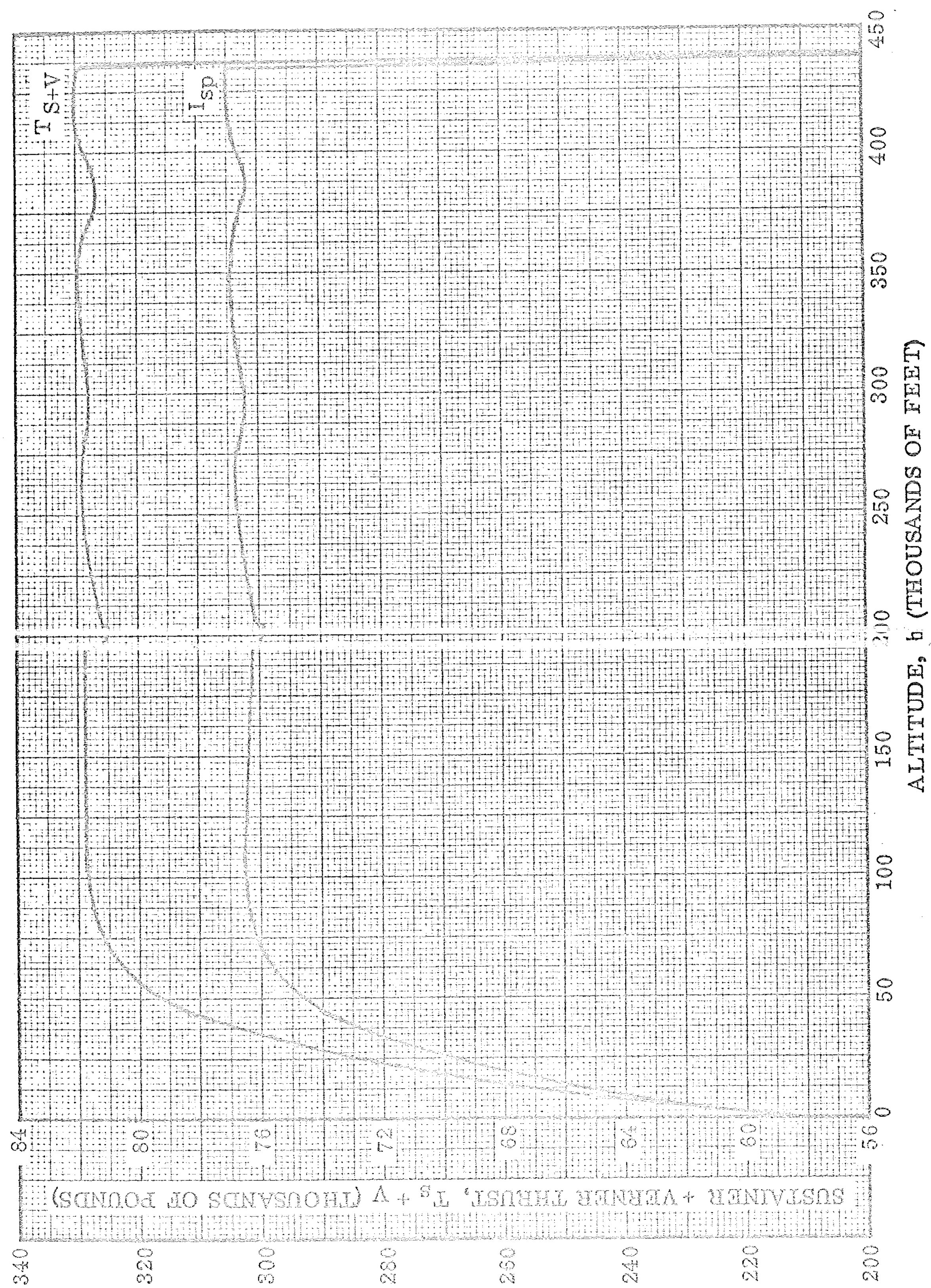


Figure 8-2. Atlas Sustainer-Plus-Vernier Thrust and Specific Impulse

8.1.1 BOOSTER ENGINE CUTOFF

The Centaur stage guidance system provides booster engine cutoff discrete, BECO. The cutoff command is issued when an axial acceleration level of 5.7g's is attained. Following a 3.1-second delay for booster engine thrust decay, the propulsion unit is jettisoned. The thrust and propellant consumption characteristics during the decay period are shown in Figure 8-3.

To preclude propellant pump cavitation, the pumps require a margin between the inlet total pressure and the liquid vapor pressure. This margin is termed net positive suction head (NPSH). The NPSH at the LO_2 sustainer pump inlet is the most critical and therefore requires special consideration. The minimum value of NPSH is reached after booster engine thrust decay, just prior to jettisoning. For AC-14 the minimum allowable NPSH is 13.0 feet (Reference 26). Figure 8-4 presents NPSH as a function of booster staging acceleration for nominal and 3-sigma dispersed conditions. It can be seen that the NPSH at the nominal staging acceleration of 5.7 g's including ± 0.1 g 3-sigma dispersion, is within allowable tolerances.

The NPSH value is calculated from the following equation:

$$NPSH = H \times F/W + \frac{144}{\rho} (P_g + P_{am} - P_{vp} - P_f) \qquad (8-1)$$

where

H	=	Liquid level, ft
F	=	Total force on liquid, lb
W	=	Weight of liquid, lb
ρ	=	Density of liquid, lb/ft^3
P_g	=	Tank pressure, lb/in^2
P_{am}	=	Ambient pressure, lb/in^2
P_{vp}	=	Vapor pressure of LO_2, lb/in^2
P_f	=	Line loss due to fluid flow, lb/in^2

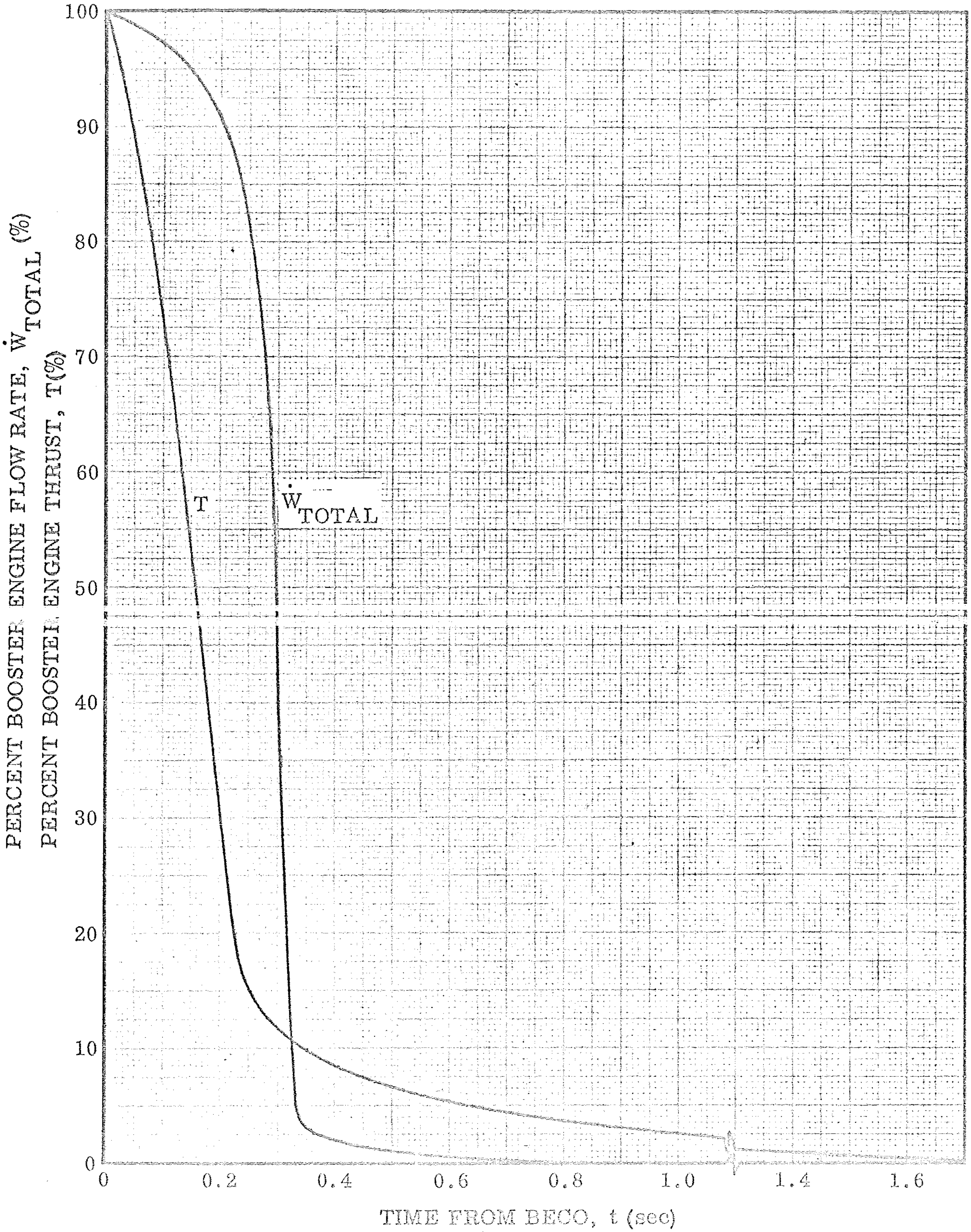


Figure 8-3. Booster Engine Decay Characteristics

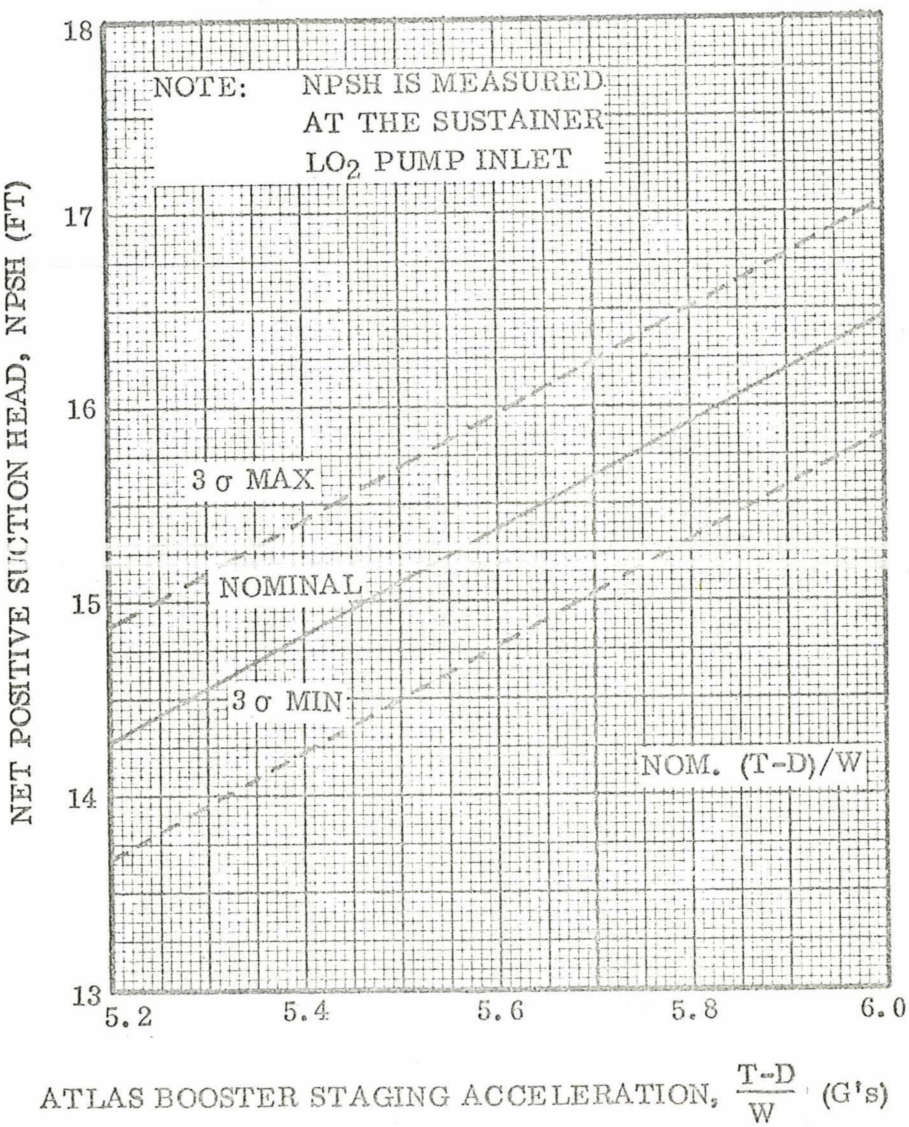


Figure 8-4. NPSH versus Booster Staging Acceleration

8.1.2 SUSTAINER ENGINE CUTOFF

Sustainer engine cutoff (SECO) is initiated at propellant depletion. A nominal sustainer engine shutdown consists of a soft phase, prior to SECO command, followed by a hard shutdown phase. The start of the soft phase begins at minimum NPSH. After minimum NPSH, the fuel manifold pressure begins to decay from its nominal operating pressure. When the pressure level falls between 600 and 700 psia, a pressure switch located in the fuel manifold is activated shutting down the sustainer and vernier engines. This constitutes SECO and initiates the hard shutdown phase. From here on, the thrust and flow rates decrease abruptly, exhibiting decay characteristics similar to that following a commanded cutoff.

In the event that the wet/dry sensor located in the fuel tank at Station 1194.24 is uncovered, a SECO discrete is issued and an immediate hard shutdown follows. This can occur either prior to or during the soft phase. This implies that fuel depletion rather than nominal LO_2 depletion has occurred. Fuel sensor locations have been chosen to minimize unburnable fuel residual. The propellant utilization system has been designed so that fuel depletion is unlikely for Surveyor-type missions.

Figure 8-5 illustrates the thrust and propellant flow characteristics for a nominal soft shutdown phase (Reference 11). Time is measured from minimum NPSH. This is the reference point from which Rocketdyne, the engine manufacturer, evaluates the engine system shutdown characteristics. The thrust and propellant consumption characteristics during the hard shutdown phase (following SECO) are also shown in Figure 8-5. The shutdown characteristics shown in Figure 8-5 are provided by Rocketdyne.

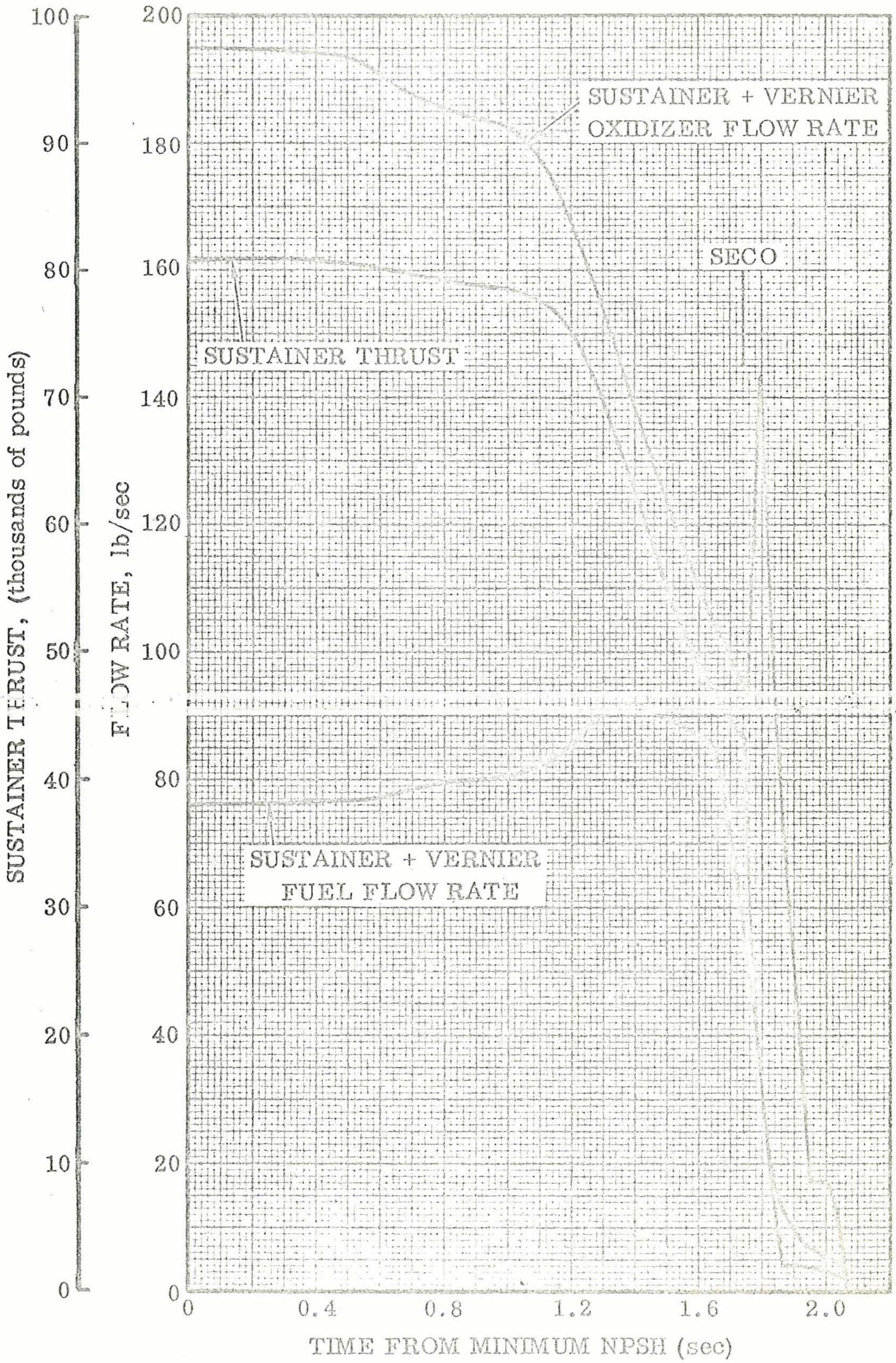


Figure 8-5. Sustainer Engine Shutdown Decay Characteristics

8.2 CENTAUR PROPULSION MODEL

Centaur stage main impulse is provided by two Pratt and Whitney (P&W) RL10A-3-3 gimbal-mounted rocket engines. The engines burn a propellant combination of liquid oxygen and liquid hydrogen. In addition, a hydrogen peroxide low-thrust propulsion system is provided for attitude and propellant level control during the coast phases of flight.

8.2.1 MAIN IMPULSE ENGINES

The Centaur propulsion characteristics are summarized in Table 8-2. The P&W engine data are based on information presented in Reference 17. The engines are calibrated to operate at a nominal oxidizer-to-fuel mixture ratio of 5:1. The nominal specific impulse of the engines at that mixture ratio is 442.87 seconds[1]. It should be recognized that although the nominal mixture ratio is 5:1, the actual mixture ratio (hence specific impulse and thrust) is affected by the propellant utilization (PU) system null setting which in turn is a function of desired propellant biases and propellant tanking levels.

Figure 8-6 shows Centaur engine thrust, specific impulse, and flow rate as a function of mixture ratio.

The propellant feed system employs a set of boost pumps on both the oxidizer and fuel sides to provide adequate pump inlet pressures for the main propellant pumps. The boost pumps are driven by hydrogen peroxide turbines. Exhaust from the turbines is axially directed and it produces an incremental force on the vehicle. The boost pump operation characteristics that affect trajectory parameters are included in Table 8-2.

[1] Engine log data.

Table 8-2. Centaur Propulsion Characteristics[1]

	NOMINAL[2]	3-σ Dispersion
Main Engines		
Centaur Vac. Thrust	30,050 lb	± 425 lb
Centaur Vac. Specific Impulse	442.54 sec	± 3.54 sec
Mixture Ratio	5.076[3]	
Boost Pumps		
Vac. Thrust	7.49 lb	± 10 %
H_2O_2 Consumption Rate	0.086 lb/sec	± 10 %

The P&W engine starts are preceded by prestart phases consisting of the boost pump operation and engine chilldown. Information pertinent to the simulation of prestart phases is shown in Table 8-3.

Table 8-3. Prestart Characteristics

	FIRST BURN	SECOND BURN
Boost Pump Operating Time	t = 44.6 sec	t = 28 sec
Chilldown Time	t = 8 sec	t = 17 sec
Chilldown Propellant Consumption	LH_2 = 23 lb LO_2 = 27 lb	LH_2 = 42 lb LO_2 = 50 lb

[1] Effective for two engines.

[2] Engine log data.

[3] Equal to main impulse oxidizer to fuel weight ratio. Used in trajectory simulations to approximate PU control.

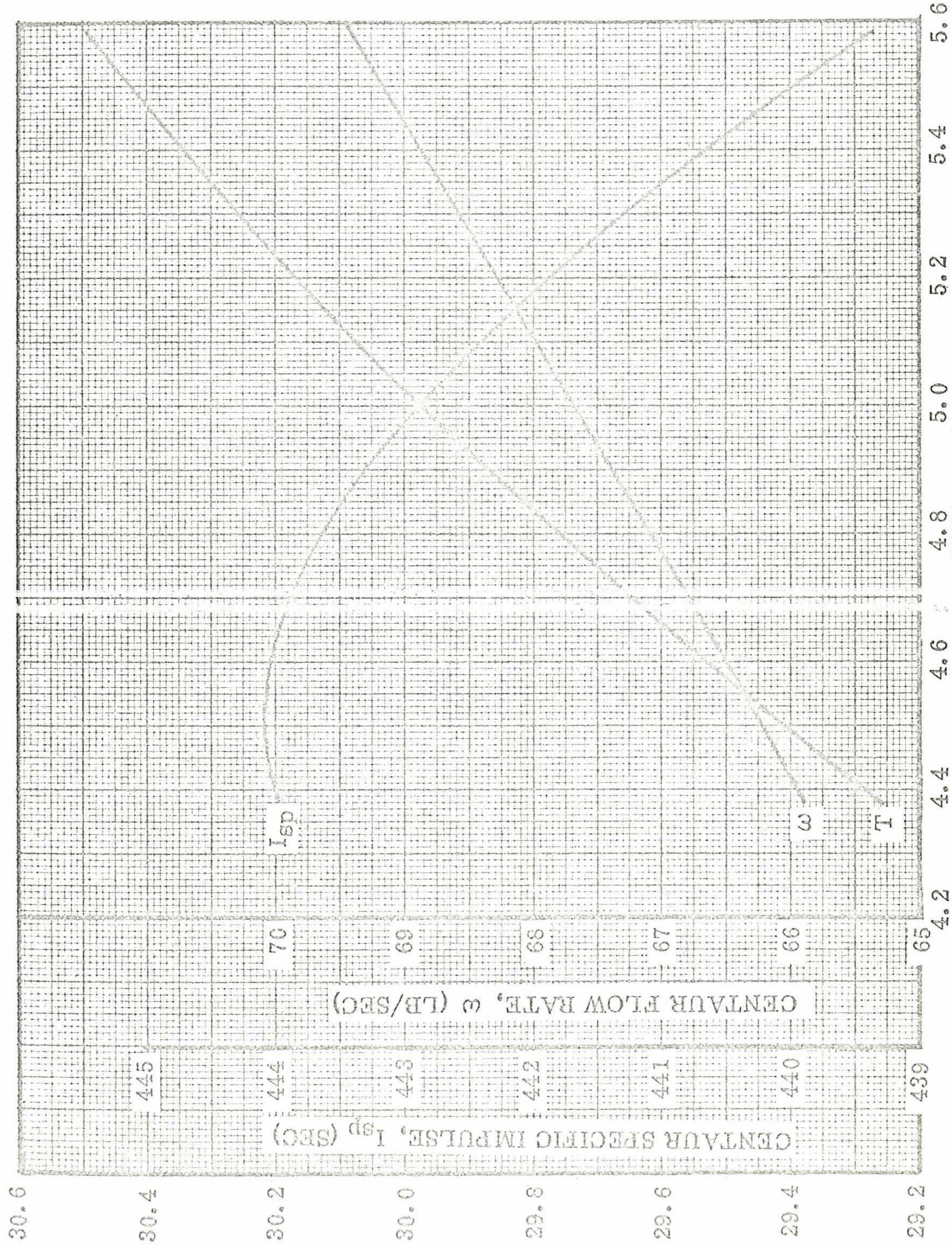


Figure 8-6. Centaur Propulsion Characteristics

Table 8-4 presents the Centaur main engine start and shutdown characteristics.

Table 8-4. Start and Shutdown Characteristics

	FIRST BURN		SECOND BURN	
	START[1]	SHUTDOWN[2]	START[1]	SHUTDOWN[2]
Total Impulse (lb-sec)	$20,000^{+6131}_{-5964}$	3110 ± 170	$19,275^{+6099}_{-5916}$	3110 ± 170
Propellant Consumption				
LH_2 (lb)	12 ± 1.0	6	12 ± 1.0	6
LO_2 (lb)	45 ± 8.0	18	43 ± 8	18

[1]Nominal start impulse from ignition to +2.0 seconds.
[2]Includes timer delay of 0.0055 seconds.

8.2.2 ENGINE REGRESSION EQUATIONS

Propulsion characteristics of the P&W RL10A-3-3 engines; i.e., mixture ratio, thrust, and specific impulse (within a given range of inlet conditions); can be determined from the trim regression equation. The general form of the equation, from Reference 17, is as follows:

$$\begin{aligned} Y_r, Y_F, Y_I = \; & C_o + C_1(FPIP) + C_2(FPIT) + C_3(OPIP) \\ & + C_4(OPIT) + C_5(FPIP)^2 + C_6(OPIP)^2 + C_7(FPIT)^2 \\ & + C_8(OPIT)^2 + C_9(FPIP)\ (FPIT) + C_{10}(FPIP)(OPIT) \\ & + C_{11}(OPIP)(OPIT) + C_{12}(FPIT)(OPIT) + C_{13}(K) \\ & + C_{14}(K)^2 + C_{15}(FPIP)(K) + C_{16}(FPIT)(K) + C_{17}(OPIP)(K) \\ & + C_{18}(OPIT)(K) \end{aligned}$$

where

Y_r = Percent nominal mixture ratio = $\dfrac{\text{mixture ratio}}{\text{nominal mixture ratio}} \times 100\%$

Y_F = Percent nominal thrust = $\dfrac{\text{thrust}}{\text{nominal thrust}} \times 100\%$

Y_I = Percent nominal specific impulse = $\dfrac{\text{specific impulse}}{\text{nominal specific impulse}} \times 100\%$

FPIP = Fuel pump inlet total pressure, psia

OPIP = Oxidizer pump inlet total pressure, psia

FPIT = Fuel pump inlet total temperature, degrees Rankine

OPIT = Oxidizer pump inlet total temperature, degrees Rankine

K = Propellant utilization valve angle loss factor (See Figure 8-7)

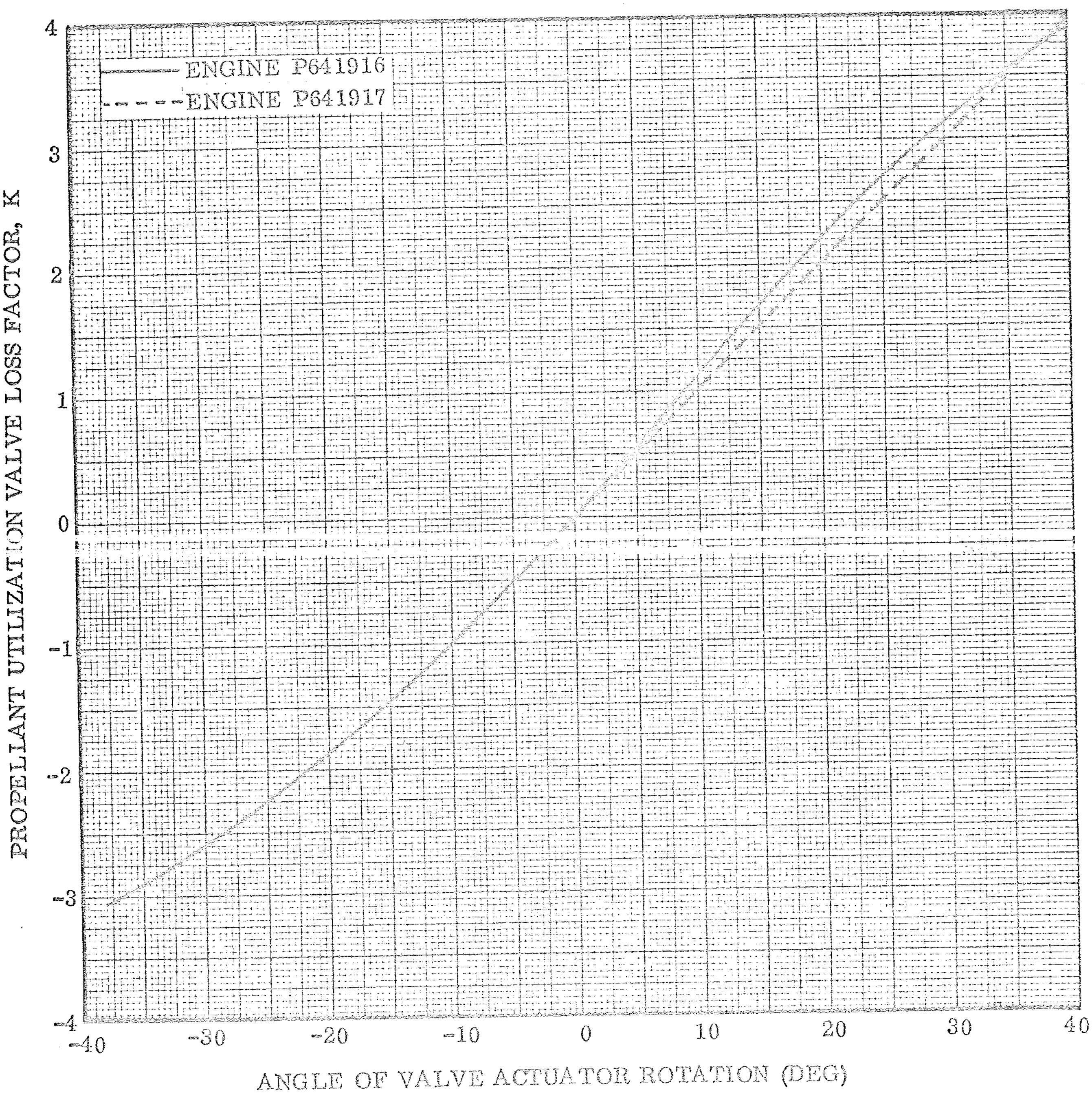


Figure 8-7. Propellant Utilization Valve Loss Factor Versus Valve Angle For AC-14 Engines

A list of constants that define the regression equations are presented in Tables 8-5 and 8-6 and can be assumed valid for the following range of inlet conditions:

PARAMETER	RANGE
FPIP	29.8 to 54.0 psia
OPIP	25.0 to 130.0 psia
FPIT	36.3 to 40.3 °R
OPIT	162.4 to 185.0 °R
K	-3.47 to +5.16

To determine the values of Y_r, Y_F, and Y_I for a specific engine, a correction must be made for the engine trim deviation and the actual engine impulse. These corrections can be made by replacing the nominal values in the regression equations by the following values:

$$\text{Mixture ratio} = \text{Nominal mixture ratio} \left[1.00 + \frac{\text{percent trim mixture ratio}}{100}\right]$$

$$\text{Thrust} = \text{Nominal thrust} \left[1.00 + \frac{\text{percent trim thrust}}{100}\right]$$

Impulse = Engine impulse from acceptance test at above mixture ratio

The nominal (rated) values are those given in Table 8-2 and obtained from Reference 17.

Table 8-5. Regression Equation Constants

CONSTANT	PERCENT MIXTURE RATIO	PERCENT THRUST	PERCENT SPECIFIC IMPULSE
C_0	45.66022	113.0937	98.14688
C_1	-0.3630909	0.2537448	0.2137699
C_2	2.813537	-0.7848150	-0.2892683
C_3	0.4693350	-0.0226822	-0.1006909
C_4	0	0	0.07840806
C_5	0	-0.0002296526	-0.0001375418
C_6	0.0001120183	-0.00006385896	-0.00009732998
C_7	0.01498950	0.007766027	-0.003763885
C_8	-0.0004508927	-0.0006488049	-0.0005027807
C_9	-0.003076895	-0.002595395	-0.001467817
C_{10}	0.001600234	-0.0008399086	-0.0007858746
C_{11}	-0.001128298	-0.0004496827	0.0005602909
C_{12}	-0.01077991	0.003366264	0.003007877
C_{15}	-0.01001839	0.002065175	-0.001764704
C_{16}	-0.05661677	0.01657675	0.03237992
C_{17}	-0.01329854	0.0007345071	0.004012892
C_{18}	0.01794036	-0.004487661	-0.009362416

Table 8-6. Variable Regression Equation Constants

CONSTANT	ENGINE	PERCENT MIXTURE RATIO	PERCENT THRUST	PERCENT SPECIFIC IMPULSE
C_{13}^{1}	P641916	-3.5491214	-0.48462962	0.36607059
	P641917	-4.2764756	-0.60731129	0.40738390
C_{14}^{1}	P641916	0.18010862	0.0032903812	-0.029038418
	P641917	0.036431548	-0.023537219	-0.023408980
C_{13}^{2}	P641916	-3.9579424	-0.58690811	0.3933233
	P641917	-4.5453445	-0.64812422	0.38050475
C_{14}^{2}	P641916	0.11004897	0.012343169	-0.031636129
	P641917	0.22550332	0.016972411	-0.028293098

[1] Use when α (Angle of valve actuator rotation) is negative.
[2] Use when α is positive.

The nominal inlet conditions for Tables 8-5 and 8-6 are the following:

PARAMETER	VALUE
FPIP	30.0 psia
OPIP	60.5 psia
FPIT	38.3 °R
OPIT	175.3 °R

8.2.3 ATTITUDE AND PROPELLANT LEVEL CONTROL ENGINES

A schematic arrangement of the attitude and propellant level control motor system is shown in Figure 8.8. The thrust direction of the A and P engines is in a plane normal to the vehicle longitudinal axis and the S and V engines are aligned with the longitudinal axis.

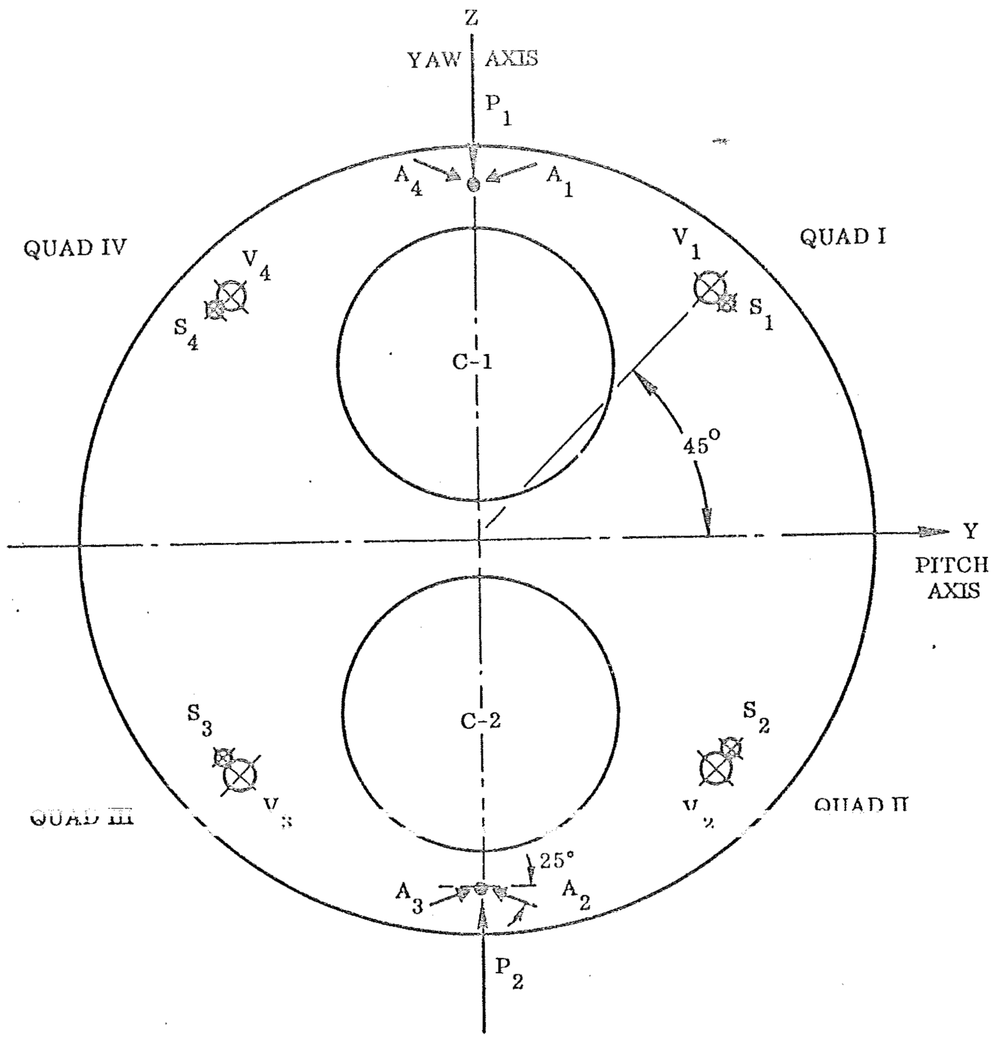
All engines use hydrogen peroxide to generate thrust. The performance characteristics of these engines are summarized in Table 8-7 and a functional description is presented in Table 8-8.

Table 8-7. Characteristics of Attitude and Propellant Level Control Engines

ENGINE DESIGNATION	THRUST (lb/engine)	SPECIFIC IMPULSE (sec)
$V_{1, 2, 3, 4}$	$52.26\ ^{+2.04}_{-2.46}$	157 ± 5
$S_{1, 2, 3, 4}$	(a) $3.07\ ^{+0.115}_{-0.140}$	155 ± 5
$P_{1, 2}$	$7.208\ ^{+0.288}_{-0.343}$	155 ± 5
$A_{1, 2, 3, 4}$	$3.582\ ^{+0.134}_{-0.164}$	155 ± 5

Table 8-8. Engine Designation and Functional Description

ENGINE DESIGNATION	FUNCTIONAL DESIGNATION	MODE OF OPERATION
$V_{1, 2, 3, 4}$	a. Propellant Settling After MECO 1 and Prior to MES 2	Continuous, Two Engines
	b. Major Re-orientation and Attitude Control	Intermittent
	c. Lateral Spacecraft/Centaur Separation Prior to Tank Blowdown Retromaneuver	Continuous, Two Engines
$S_{1, 2, 3, 4}$	a. Propellant Settling During Parking Orbit Coast Phase, Excluding Periods of V Engine Operation Defined in (a) Above	Continuous, Two Engines
$A_{1, 2, 3, 4}$ and $P_{1, 2}$	a. Attitude Control During Centaur Coast Phases	Intermittent



NOTE: ARROWS INDICATE DIRECTION OF FORCE

ENGINE	THRUST
V	≈50#
S	≈3#
A	≈3.5#
P	≈7#

Figure 8-8. Arrangement of Attitude and Propellant Level Control Engines

SECTION 9

FLIGHT CONTROL SYSTEM

The Atlas/Centaur flight control system consists of two autopilot sets and the Minneapolis-Honeywell all-inertial vehicle guidance set. The Atlas autopilot, mounted within the equipment pod of the first stage, provides programmed steering of the vehicle from launch through separation of the Atlas booster engine package. This autopilot also executes steering commands furnished by the second stage guidance set after Atlas booster staging through sustainer engine cutoff. Sequential switching operations, as required by the first stage, are provided by the first stage autopilot programmer.

The Centaur autopilot is mounted on the forward bulkhead of the second stage and accomplishes the following functions:

a. Stabilizes the Centaur vehicle during second stage flight.

b. Executes guidance system steering commands.

c. Provides switching sequences, as needed.

The all-inertial vehicle guidance set located on the Centaur stage will generate pitch and yaw steering signals to the autopilots for flight control from booster engine cutoff (BECO) plus 8 seconds to termination of the Centaur retromaneuver. The guidance set will issue the primary discrete for booster engine cutoff and Centaur main engine cutoff.

9.1 ATLAS STAGE AUTOPILOT

The Atlas stage autopilot performs the following functions:

a. Rolls the vehicle to the required launch azimuth.

b. Pitches and yaws the vehicle according to predetermined programs.

c. Performs vehicle steering from Centaur vehicle guidance set information during the sustainer solo phase of flight.

d. Generates timed sequences and switching functions as required for vehicle control.

9.2 ATLAS BOOSTER ROLL PROGRAM

The Atlas/Centaur launch pad, Complex 36B, is aligned along an azimuth of 115 degrees, and since the vehicle pitch-over maneuver is restricted to the vehicle referenced pitch plane, a roll maneuver must be executed prior to the termination of vertical rise in order to achieve the desired azimuth. AC-14 will have a launch azimuth somewhere between 78 and 115 degrees, depending on launch day and launch time. The AC-14 Firing Tables, Reference 2, should be consulted for launch azimuth and launch time relationship for each launch day. Table 9-1 presents roll maneuver data. The times are referenced to 2-inch motion. A counter-clockwise roll, when viewed from above the launcher, is defined as a positive rate.

Table 9-1. AC-14 Roll Program

TIME (sec)	ROLL RATE (deg/sec)
0 → 2	0
2 → 15	Launch-Day and Time Dependent
15 → BECO	0

9.3 ATLAS PITCH AND YAW PROGRAMS

The Altas autopilot has a basic pitch program (PP181) and a basic yaw program (YP0) built into it. To make allowance for seasonal wind changes and hence maximize launch availability, several delta pitch and delta yaw programs are available for use (see Tables 9-2 and 9-3). These delta programs are added to the basic programs by the Centaur guidance system as required. On the day of launch, wind soundings are taken to determine the required delta-pitch and delta-yaw programs to load into the Centaur guidance system. For targeting and payload reporting purposes the basic pitch and yaw programs were used, i.e., PP181 and YP0.

Table 9-2. Pitch Programs[1]

TIME (sec)	PP181 DELTA PITCH RATE (deg/sec)	PP181 TOTAL PITCH RATE[2] (deg/sec)	PP211 DELTA PITCH RATE (deg/sec)	PP211 TOTAL PITCH RATE (deg/sec)	PP141 DELTA PITCH RATE (deg/sec)	PP141 TOTAL PITCH RATE (deg/sec)	PP101 DELTA PITCH RATE (deg/sec)	PP101 TOTAL PITCH RATE (deg/sec)	PP31 DELTA PITCH RATE (deg/sec)	PP31 TOTAL PITCH RATE (deg/sec)
0-15	0	0	0	0	0	0	0	0	0	0
15-25	0	0.51	+0.04	0.55	-0.01	0.50	-0.18	0.33	-0.31	0.20
25-45	0	0.54	+0.01	0.55	-0.03	0.51	+0.02	0.56	+0.01	0.55
45-50	0	0.63	-0.06	0.57	+0.04	0.67	+0.02	0.65	+0.08	0.71
50-60	0	0.66	-0.03	0.63	-0.01	0.65	+0.04	0.70	+0.09	0.75
60-75	0	0.60	-0.03	0.57	+0.07	0.67	+0.08	0.68	+0.14	0.74
75-79	0	0.57	-0.05	0.52	+0.07	0.64	+0.06	0.63	+0.15	0.72
79-87	0	0.84	+0.03	0.87	-0.09	0.75	-0.04	0.80	-0.08	0.76
87-103	0	0.69	+0.01	0.70	-0.01	0.68	-0.04	0.65	-0.06	0.63
103-130	0	0.36	0	0.36	0	0.36	+0.01	0.37	+0.03	0.39
130-BECO	0	0.24	0	0.24	0	0.24	-0.01	0.23	-0.01	0.23

[1] Sign convention: A positive (+) rate turns the vehicle nose down whereas a negative (-) rate turns it nose up.

[2] The basic pitch program.

Table 9-3. Yaw Programs[1]

TIME (sec)	YP0[2] DELTA YAW RATE (deg/sec)	YP6 DELTA YAW RATE (deg/sec)	YP7 DELTA YAW RATE (deg/sec)
0→15	0	0	0
15→25	0	0	0
25→45	0	+0.15	+0.02
45→50	0	+0.03	-0.20
50→60	0	-0.12	+0.07
60→75	0	0	0
75→79	0	-0.10	+0:10
79→87	0	-0.06	+0.08
87→103	0	+0.04	+0.04
103→130	0	+0.03	+0.04
130→BECO	0	0	-0.01

1 Sign convention: A positive (+) rate turns the vehicle nose left whereas a negative (-) rate turns it nose right.

2 The basic yaw program.

SECTION 10

WEIGHTS

10.1 WEIGHT DATA

A weight breakdown for the AC-14 launch configuration is presented in Tables 10-1 through 10-3. These weights were obtained from Reference 4 and reflect the configuration status as of 21 August 1967.

Table 10-1. AC-14 Launch Configuration Weight Summary

CONFIGURATION	WEIGHT INCREMENTS (lb)	AT ATLAS IGNITION	AT 2-INCH MOTION
SPACECRAFT SEPARATED*		2,223	2,223
CENTAUR TANKED WEIGHT		37,976	
Less Ground Boiloff Weight	68		
CENTAUR GROSS LAUNCH WEIGHT			37,908
ATLAS TANKED WEIGHT		287,494	
Less Ground Expendables and Pre-ignition GO_2 Loss	3,006		
ATLAS GROSS LAUNCH WEIGHT			284,488
COMBINED VEHICLE GROSS WEIGHT INCLUDING SPACECRAFT		327,693	324,619

*See Reference 27.

Table 10-2. Centaur Stage Weight Summary

ITEM NO.	DESCRIPTION	WEIGHT (lb)
	CENTAUR TANKED WEIGHTS*	37,976
1	Centaur Jettison Weights*	4,111
1.1	Centaur Dry Weights*	3,629
1.1.1	Basic Vehicle Weight	(3,231)
	Body Group	974
	Basic Structure	878
	Secondary Structure	96
	Propulsion Group	1,223
	Main Engines	593
	Fuel System	215
	Oxidizer System	254
	PLIS and PU System	95
	H_2O_2 System	66
	Guidance Group	333
	Guidance System	226
	Autopilot System	107
	Control Group	152
	Hydraulic System	91
	Attitude Control System	26
	Ullage Motors	35
	Pressurization Group	187
	Electrical Group	283
	Separation Equipment	79
1.1.2	Useful Load*	(398)
	Flight Instrumentation	255
	Range Safety System	51
	Azusa Tracking System	0
	C-Band Tracking System	25
	TLM RF System No. 2	0
	TLM RF System No. 1	121
	Guidance Signal Conditioner	11
	P&WA Instrumentation Boxes	28
	Landline Systems	9
	General TLM Equipment	10
	Miscellaneous Equipment	143
	TLM RF System No. 4	0

*Does not include spacecraft weight.

Table 10-2. Centaur Stage Weight Summary, Contd

ITEM NO.	DESCRIPTION	WEIGHT (lb)
	Destruct Installation - Surveyor	15
	Spectrometer Installation	0
	Spacecraft Adapter and Equipment	128
1.2	Residuals	482
1.2.1	Propellants	(422)
	LH_2 Trapped	60
	LO_2 Trapped	54
	LH_2 PU Residual	20
	LO_2 PU Residual	0
	Gaseous H_2	116
	Gaseous O_2	172
1.2.2	Hydrogen Peroxide	(41)
	Retromaneuver - Attitude Control	23
	Trapped	5
	Reserve	13
	Contingency	0
1.2.3	Helium	(7)
1.2.4	Ice	(12)
2	Centaur Expendables	30,528
2.1	Propellants	30,330
2.1.1	Main Impulse	(29,856)
	H_2	4,914
	O_2	24,942
2.1.2	Gas Boiloff on Ground[1]	(68)
	H_2	27
	O_2	41
2.1.3	Inflight Chilldown	(142)
	H_2	65
	O_2	77
2.1.4	Boost Phase Vent	(119)
	H_2	53
	O_2	66
2.1.5	Sustainer Phase Vent	(90)
	H_2	30
	O_2	60

[1] Expended prior to Atlas stage ignition.

Table 10-2. Centaur Stage Weight Summary, Contd

ITEM NO.	DESCRIPTION	WEIGHT (lb)
2.1.6	Engine Decay - First Burn	(24)
	H_2	6
	O_2	18
	Engine Decay - 2nd Burn	(24)
	H_2	6
	O_2	18
2.1.7	Parking Orbit Vent	(5)
	H_2	5
	O_2	0
2.1.8	Parking Orbit Leakage	(2)
	H_2	0
	O_2	2
2.2	Hydrogen Peroxide	196
2.2.1	Boost Pumps	(43)
2.2.2	Ullage + Control Motors	(153)
2.3	Helium Expended	2
3	Equipment Jettisoned in Boost Phase	3,337
3.1	Nose Fairing	2,046
3.2	Insulation Panels	1,241
3.3	Ablated Ice	50

Table 10-3. Atlas Stage Weight Summary

ITEM NO.	DESCRIPTION	WEIGHT (lb)
	ATLAS TANKED WEIGHTS	287,494
1	Sustainer Jettison Weight	8,821
1.1	Sustainer Dry Weight	6,110
1.2	Sustainer Residuals	1,641
1.3	Unburned Expendables	0
1.4	Interstage Adapter	1,054
1.5	Unburned Lube Oil	16
2	Booster Jettison Weight	7,484
2.1	Booster Dry Weight	6,318
2.2	Booster Residuals	1,129
2.3	Unburned Lube Oil	37
3	Atlas Expendables	271,189
3.1	Flight Expendables	268,183
3.1.1	Main Impulse	(267,977)
	RP-1	83,123
	O_2	184,854
3.1.2	Helium-Panel Purge	(6)
3.1.3	Oxidizer Vent Loss	(15)
3.1.4	Lube Oil	(185)
3.2	Ground Expendables*	2,556
3.2.1	Fuel	(540)
3.2.2	Oxidizer	(1,709)
3.2.3	Lube Oil	(3)
3.2.4	Exterior Ice	(54)
3.2.5	LN_2 in Helium Shrouds	(250)
3.3	Preignition GO_2 Loss	450

* 2.06 seconds ground run time.

10.2 PROPELLANT LOADING

10.2.1 ATLAS PROPELLANT LOADING

Atlas propellant loading calculations are summarized in Table 10-4. The main impulse propellants are calculated by subtraction of the unusable propellants from the initial tanked quantities.

Table 10-4. Atlas Propellant Loading Calculations

ITEM	FUEL	OXIDIZER
1. Total Tank Volume	1,705.84 ft^3	2,726.22 ft^3
2. Line Volumes	8.00 ft^3	22.83 ft^3
3. Total Available Volume, (1) + (2)	1,713.84 ft^3	2,749.05 ft^3
4. Ullage Volume	19.69 ft^3	28.65 ft^3
5. Net Tank Volume, (3) - (4)	1,694.15 ft^3	2,720.40 ft^3
6. Propellant Density	50.00 lb/ft^3	69.30 lb/ft^3[1]
7. Net Tanked Propellants, (5) × (6)	84,708 lb	188,524 lb
8. Non-impulse Propellants	1,178 lb	2,857 lb
a. Ground Main Tank Vent	0 lb	450 lb
b. Ground Run	540 lb	1,709 lb
c. Inflight Vent	0 lb	15 lb
d. Booster Engine Liquid Residuals	487 lb	554 lb
e. Sustainer Engine Liquid Residuals	94 lb	55 lb
f. Vernier Engine Liquid Residuals	57 lb	74 lb
9. Propellants Contributing to NPSH at Liftoff (7) - (8)	83,530 lb	185,667 lb
10. Vaporized Liquid	0 lb	474 lb
Net NPSH Expendables at Liftoff (9) - (10)	83,530 lb	185,193 lb

[1]At 25.0 psia Phase II pressure

10.2.2 CENTAUR PROPELLANT LOADING

Centaur propellant loading calculations are summarized in Table 10-5. The main impulse propellants are calculated by subtraction of the unusable propellants from the initial tanked quantities.

Table 10-5. Centaur Propellant Loading Calculations

ITEM	FUEL	OXIDIZER
1. Total Tank Volume	1,265.40 ft^3	375.71 ft^3
2. Line Volumes	2.53 ft^3	1.85 ft^3
3. Total Available Volume, (1) + (2)	1,267.93 ft^3	377.50 ft^3
4. Ullage Volume	11.23 ft^3	6.61 ft^3
5. Net Tank Volume, (3) - (4)	1,256.70 ft^3	370.95 ft^3
6. Propellant Density	4.218 lb/ft^3 [1]	68.60 lb/ft^3 [2]
7. Net Tanked Propellants, (5) × (6)	5,301 lb	25,447 lb
8. Non-impulse Propellants	387 lb	505 lb
a. Ground Vent	27 lb	41 lb
b. Boost Phase Vent	53 lb	66 lb
c. Sustainer Phase Vent	30 lb	60 lb
d. Coast Phase Vent	45 lb	0 lb
e. Coast Phase Leakage	0 lb	2 lb
f. Engine Chill	65 lb	77 lb
g. Vaporized Liquid	115 lb	169 lb
h. Trapped Liquid	60 lb	62 lb
i. PU System Bias	20 lb	0 lb
j. Engine Shutdown Loss	12 lb	36 lb
9. Net Main Impulse, (7) - (8)	4,914 lb	24,942 lb

[1] At 20.25 psia tanking pressure

[2] At 30.50 psia tanking pressure

SECTION 11
AERODYNAMICS

The gross vehicle aerodynamic coefficients applicable to the Atlas/Centaur are documented in Reference 18. These data are based primarily on results of six-component force and moment tests of 1/35 scale models conducted at the Convair high-speed wind tunnel. Testing was conducted for Mach numbers between 0.62 and 4.36 which covers the period of flight when aerodynamic forces and moments are greatest. To cover the complete Mach number range, the test data were extrapolated by empirical and theoretical methods.

The coefficient of total vehicle aerodynamic force is normally resolved into three mutually orthogonal components that are aligned axially and laterally with the vehicle longitudinal axis. From a performance viewpoint, however, the two most significant components of force are the axial (drag) and the normal (pitch-plane) force. The axial force is defined as positive when it acts to retard the forward motion of the vehicle. Normal force is defined as positive when it acts in the upward sense.

11.1 AXIAL FORCE

During the booster phase, the drag model for Atlas/Centaur divides the total aerodynamic axial force into a component that is dependent on ambient dynamic pressure and a component that varies linearly with the ratio of ambient static pressure to sea level ambient pressure. A time-varying force to account for launcher holddown and base suction has also been incorporated; this retarding axial force is applied during the first 10 seconds of flight (measured from liftoff).

During booster phase of flight, total aerodynamic axial force is calculated by means of the following equations:

$$\text{Axial force} = CA(q)(S) - 4500\,(1 - P/P_o) + H \qquad (11\text{-}1)$$

for $t \le 10$ seconds, and

$$\text{Axial force} = CA(q)(S) - 4500\,(1 - P/P_o) \qquad (11\text{-}2)$$

for $t > 10$ seconds,

where CA = q-dependent axial force coefficient

q = Free-stream dynamic pressure, psf

S = Reference area, 78.5 ft^2

P/P_o = Ratio of ambient to sea-level static pressure, psf/psf

t = Flight time

H = Combined holddown and base suction force, lb

During sustainer phase of flight, total axial force is simply

$$\text{Axial force} = CA(q)(S) \qquad (11\text{-}3)$$

CA is presented versus Mach number in Figure 11-1. This coefficient reflects changes to the original Atlas/Centaur base drag component of axial force.

The term 4500 $(1-P/P_o)$ in Equations 11-1 and -2 is a booster-phase, base force component of axial force that is a function of free-stream static pressure, where

4500 = Vacuum base thrust, lb

P/P_o = Ratio of ambient to sea-level static pressure, psf/psf

The holddown and base suction force, H, is plotted in Figure 11-2 as a function of time from liftoff. The analytical expression for this force is

$$H = \text{Retarding axial force} = 41.290\,(10 - t)^{2.3917} \tag{11-4}$$

where $t \leq 10$ seconds of flight,

H = Retarding axial force, lb.

Implicit in the above discussion is the assumption that the total angle of attack, α_T, is very small. This assumption is justified since axial force is essentially constant with angle of attack (within the permissible range of angles of attack for Atlas/Centaur vehicles).

For a detailed discussion of the new Atlas/Centaur drag model see Reference 19.

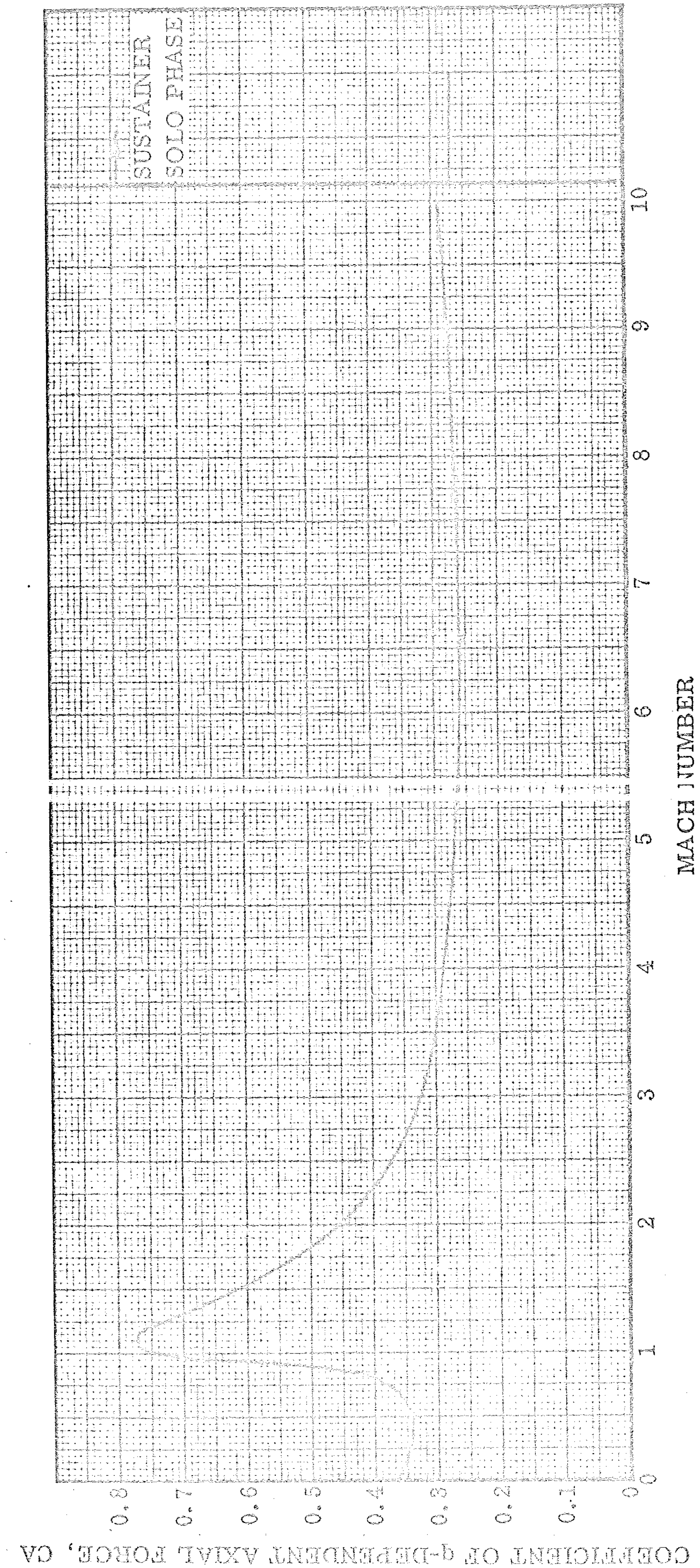


Figure 11-1. Atlas/Centaur Axial Force Coefficient (q-Dependent Component of Total Axial Force)

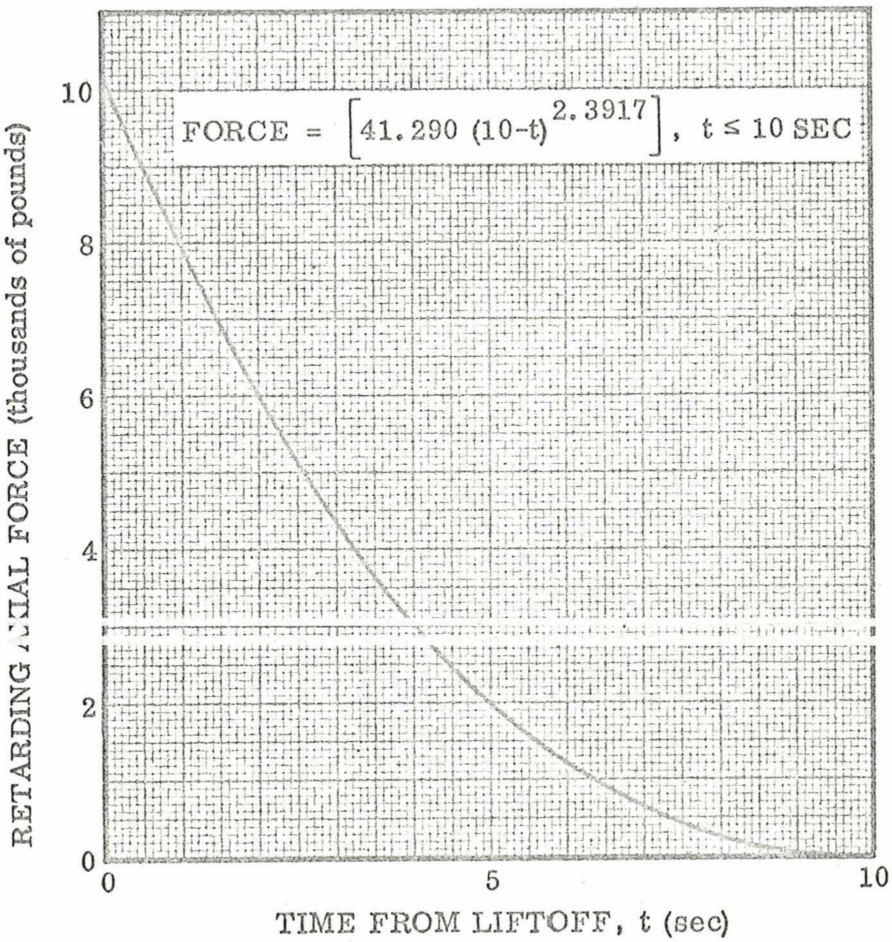


Figure 11-2. Atlas/Centaur Holddown and Base Suction Force

11.2 NORMAL FORCE

Normal force coefficient is presented in two components, viz., the component at zero angle of attack, CNO, and the component that varies with angle of attack, CN^*/α. The total normal force is the sum of the two components:

$$N = [CNO + (CN^*/\alpha)\alpha]\, qS \qquad (11\text{-}5)$$

where α = Pitch-plane angle of attack, deg

q = Free-stream dynamic pressure, psf

S = Reference area, 78.5 ft^2

N = Normal force, lb

CNO is plotted in Figure 11-3 as a function of Mach number. CN^*/α, per degree α, is presented versus Mach number in Figures 11-4 and 11-5.

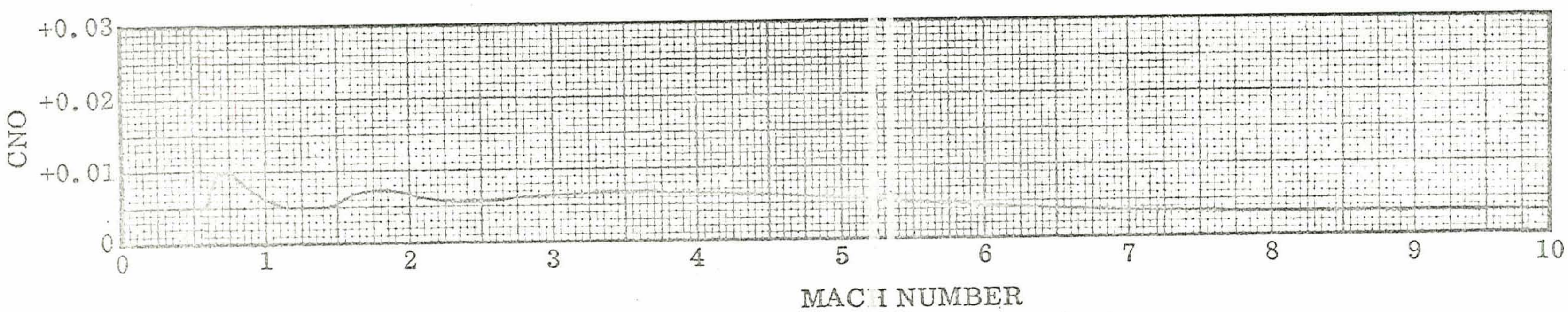


Figure 11-3. Total Vehicle Normal Force at $\alpha = 0$

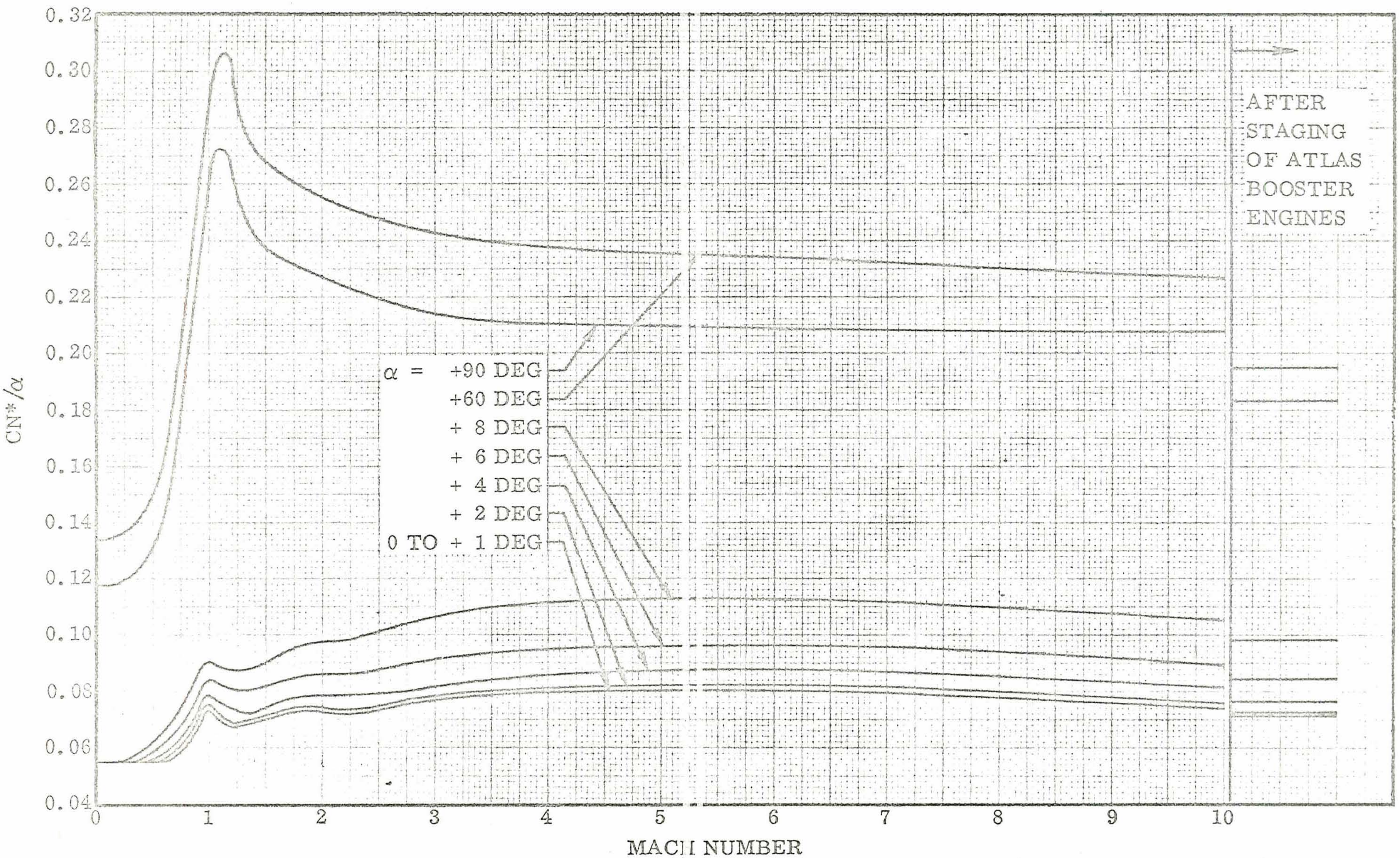


Figure 11-4. Component of Total Vehicle Normal Force Coefficient Due to Angle of Attack, Positive α Only

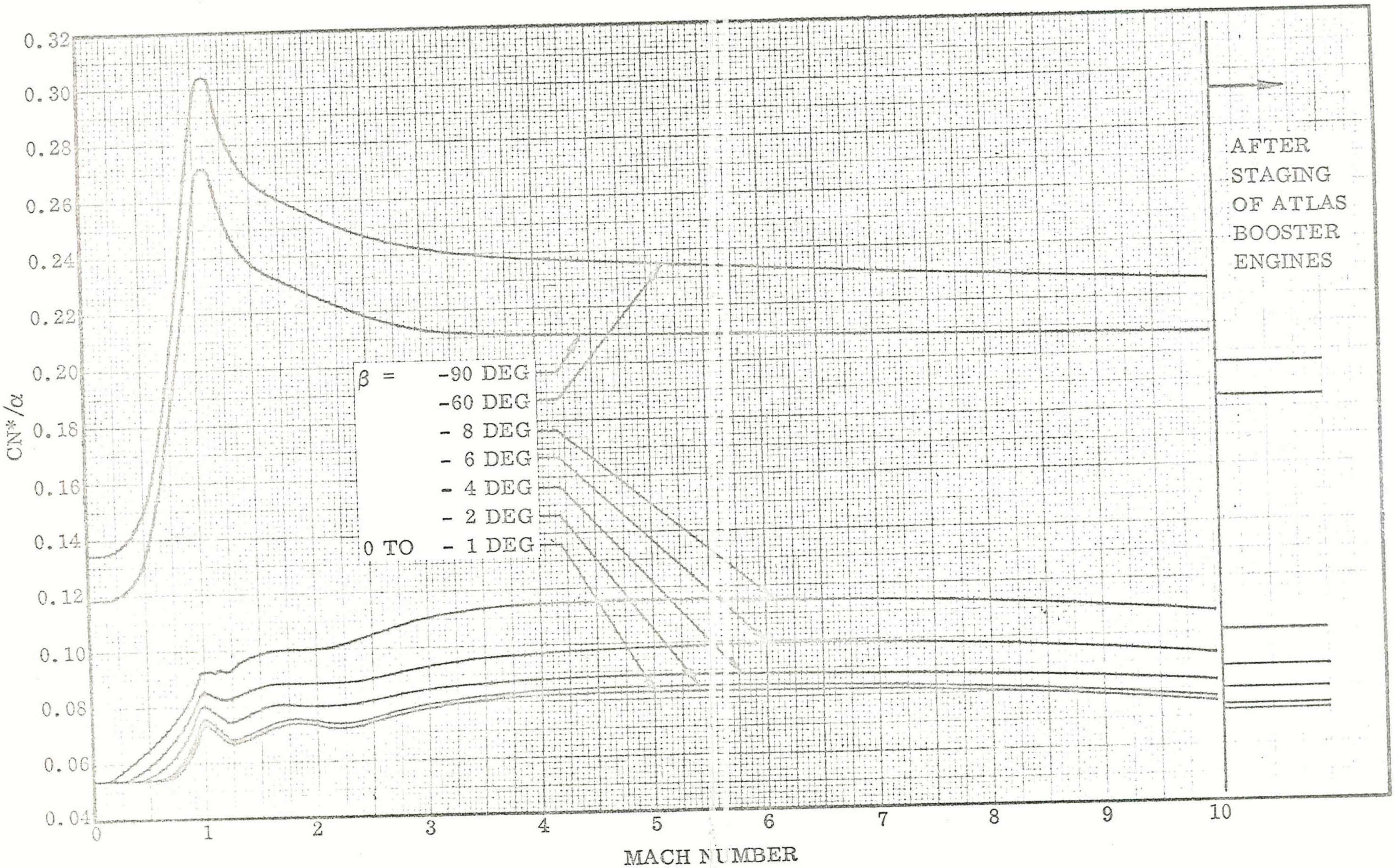


Figure 11-5. Component of Total Vehicle Normal Force Coefficient Due to Angle of Attack, Negative α Only

11.3 PITCHING MOMENT COEFFICIENTS

Pitching moment is positive for a moment tending to rotate the nose of the vehicle upward. The total pitching moment MN is defined by the equation

$$MN = CMNO(q)(S)(LREF) - N(XCP/LREF)(LREF) \tag{11-6}$$

where CMNO = Pitching moment coefficient at N=0

q = Free-stream dynamic pressure, psf

S = Reference area, 78.5 ft^2

N = Total normal force, lb

LREF = Reference length, 1500 in.

XCP/LREF = Center of pressure of normal force

$$= \frac{\text{(pitching moment at N=0) - (total pitching moment)}}{\text{(normal force) (LREF)}} \tag{11-7}$$

The effective center of pressure in the pitch plane is therefore

$$X_{CP} = \left(\frac{MN}{N}\right) \tag{11-8}$$

inches aft of Atlas Station zero.

The pitching moment coefficient (CMNO) is presented in Figure 11-6. The XCP/LREF is presented versus Mach number in Figures 11-7 and 11-8.

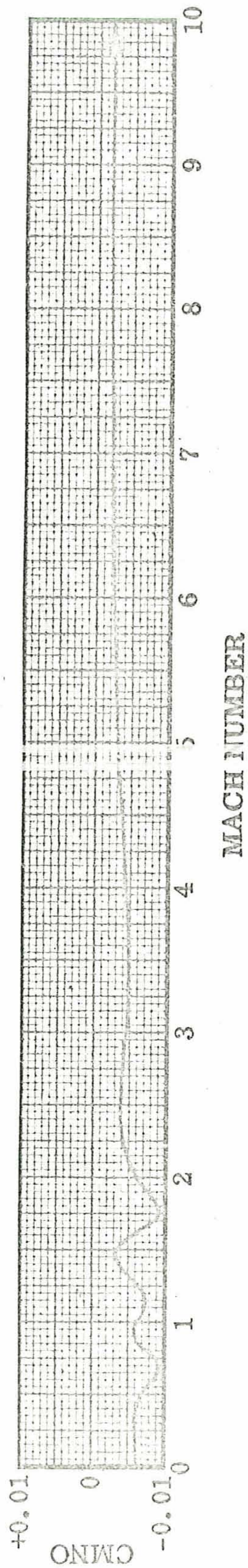


Figure 11-6. Pitching Moment Coefficient at CN = 0

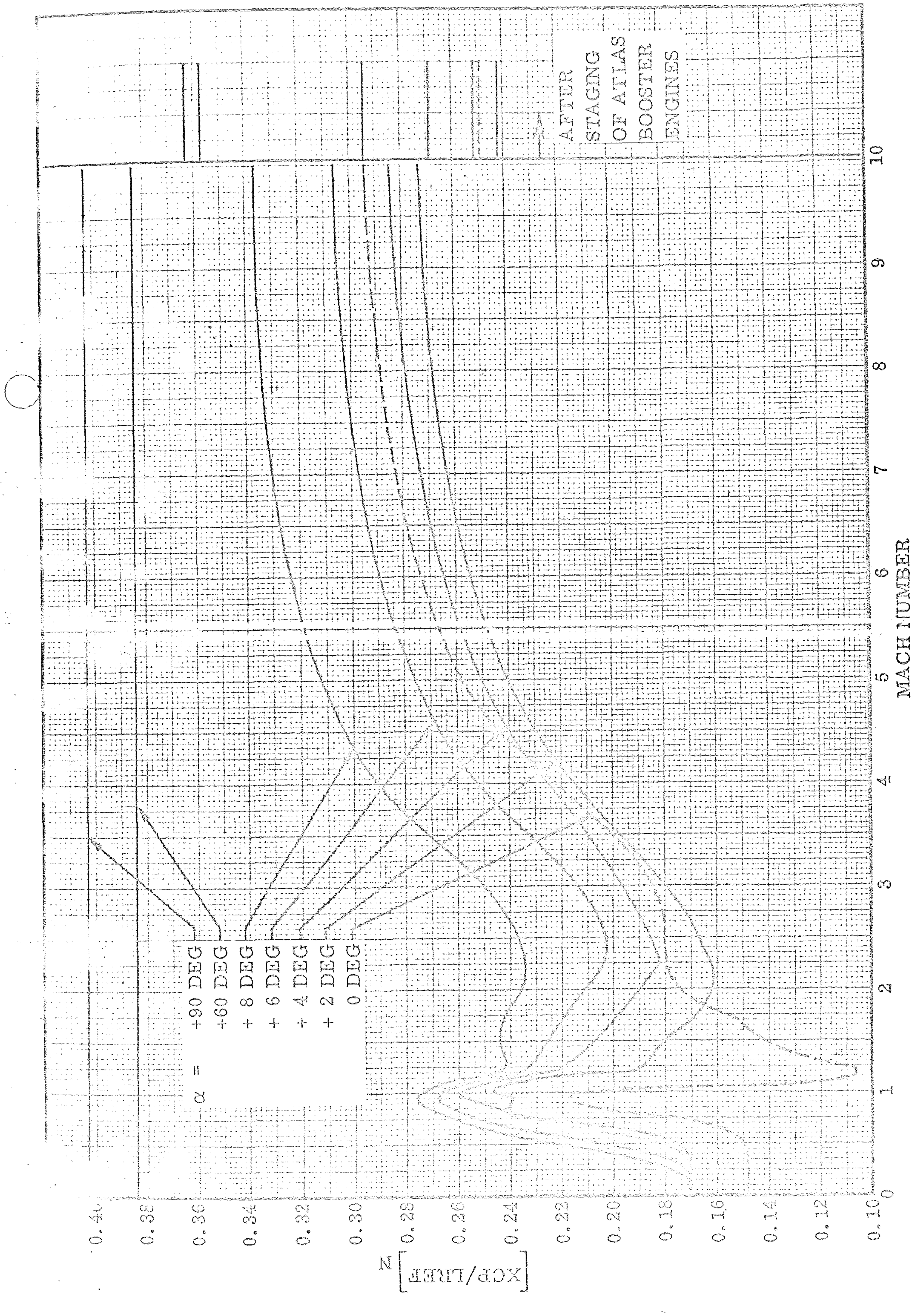


Figure 11-7. Center of Pressure of Total Vehicle Normal Force Component, Positive α Only

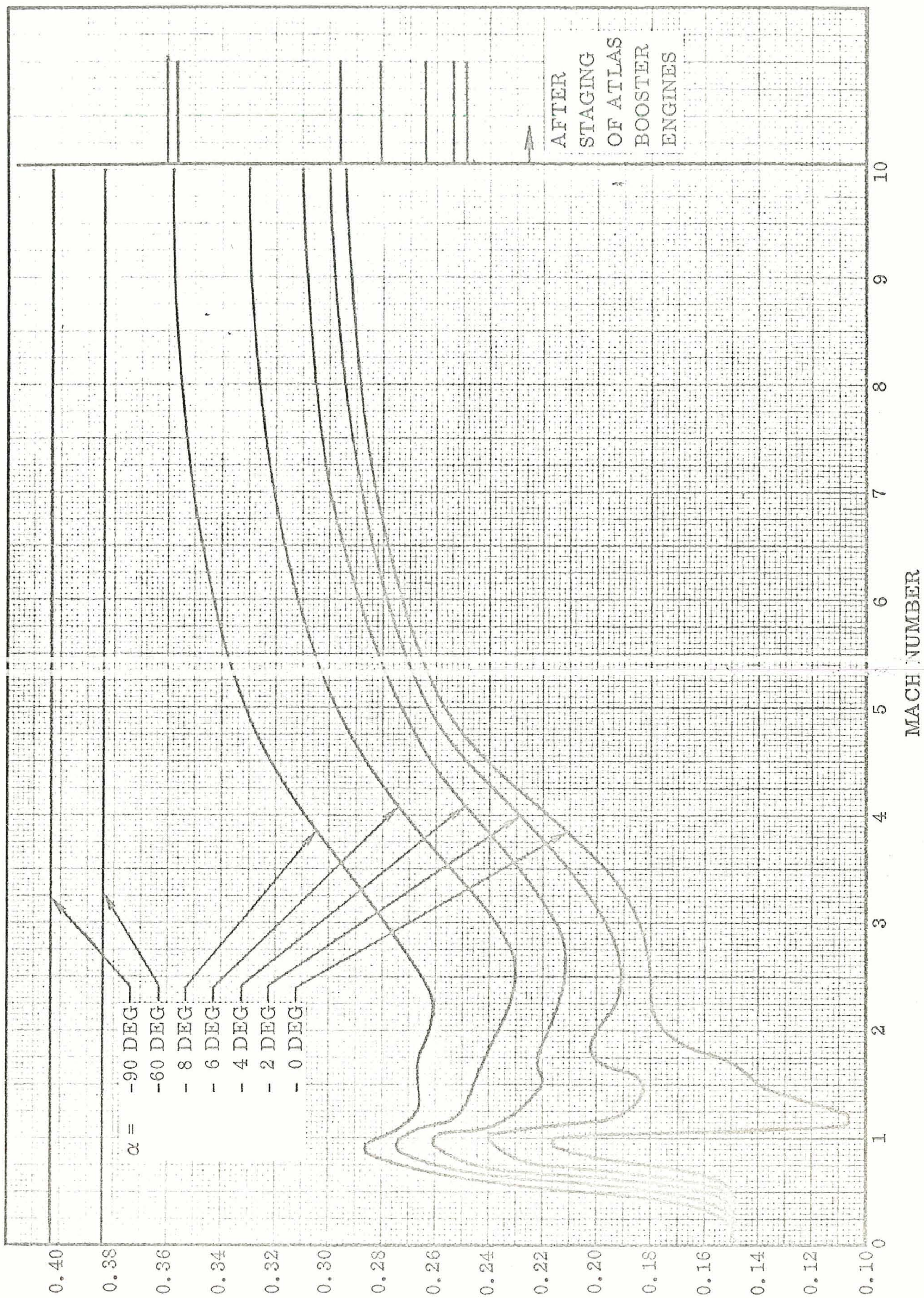


Figure 11-8. Center of Pressure of Total Vehicle Normal Force Component, Negative α Only

SECTION 12

PHYSICAL AND ATMOSPHERIC DATA

This section presents the physical and atmospheric data used in generation of earth-moon trajectory simulations.

12.1 EARTH DATA

12.1.1 EARTH MODEL

The geometric form of the earth assumed in trajectory simulations is the Fischer earth model (1960), Reference 21. Constants defining the earth model are:

Equatorial earth radius,

$$A = 6,378.166000 \text{ kilometers (exact)}$$

$$= 20,925,741.47 \text{ international feet}$$

Flattening,

$$f = \text{Flattening} = 1 - \frac{B}{A} = 1/298.30$$

Polar earth radius,

$$B = 6,356.784284 \text{ kilometers}$$

$$= 20,855,591.48 \text{ international feet.}$$

12.1.2 EARTH ROTATION RATE

The earth rotation rate (ω_E) assumed, Reference 21, is given by

$$\omega_E = \frac{360}{86164.09892 + 0.00164T} \text{ deg/sec}$$

Where T is the number of Julian centuries of 36,525 days from 1900 January 0.5 U.T. (Julian date = 2,415,020.0). For the year 1963,

$$\omega_E = 4.1780742 \times 10^{-3} \text{ deg/sec}$$

12.1.3 EARTH POTENTIAL FUNCTION

The potential function of the earth assumed, Reference 21, is given by

$$\phi(R,\phi) = \frac{GM_E}{R}\left[1 + \frac{JR_E^2}{3R^2}(1 - 3\sin^2\phi) + \frac{H}{5}\frac{R_E^3}{R^3}(3 - 5\sin^2\phi)\sin\phi\right.$$

$$\left. + \frac{D}{35}\frac{R_E^4}{R^4}(3 - 30\sin^2\phi + 35\sin^4\phi)\right]$$

where,

GM_E = $3.986032\ (\pm 0.000030) \times 10^5\ km^3/sec^2$

= $1.4076539 \times 10^{16}\ ft^3/sec^2$

= $62{,}750.59\ n.mi.^3/sec^2$

R_E = Equatorial earth radius

R = Geocentric radius

ϕ = Geocentric latitude

J = $1.62345\ (\pm 0.00030) \times 10^{-3}$

H = $-0.575\ (\pm 0.025) \times 10^{-5}$

D = $0.7875\ (\pm 0.0875) \times 10^{-5}$

12.1.4 LAUNCH PAD COORDINATES

The launch pad coordinates for ETR pads 36A and 36B, referenced to the earth model described in Section 12.1.1, are:

PAD	GEOCENTRIC LATITUDE (deg)	WEST LONGITUDE (deg)	GEODETIC TIP ANGLE (deg)
36A	28.310551	80.537949	0.160960
36B	28.307492	80.541179	0.160947

12.1.5 ATMOSPHERIC PROPERTIES

The atmospheric model used in trajectory simulations is based on data from the Patrick Reference Atmosphere and the 1959 ARCD Model Atmosphere given in References 22 and 23. Equations relating pressure, density, and temperature to altitude are presented in Reference 24.

12.1.6 WIND DATA

ETR wind profiles are presented in Figures 12-1 through 12-13. The wind profiles represent mean monthly statistical wind distribution measured at Patrick Air Force Base, Florida. The wind profiles are documented in Reference 25.

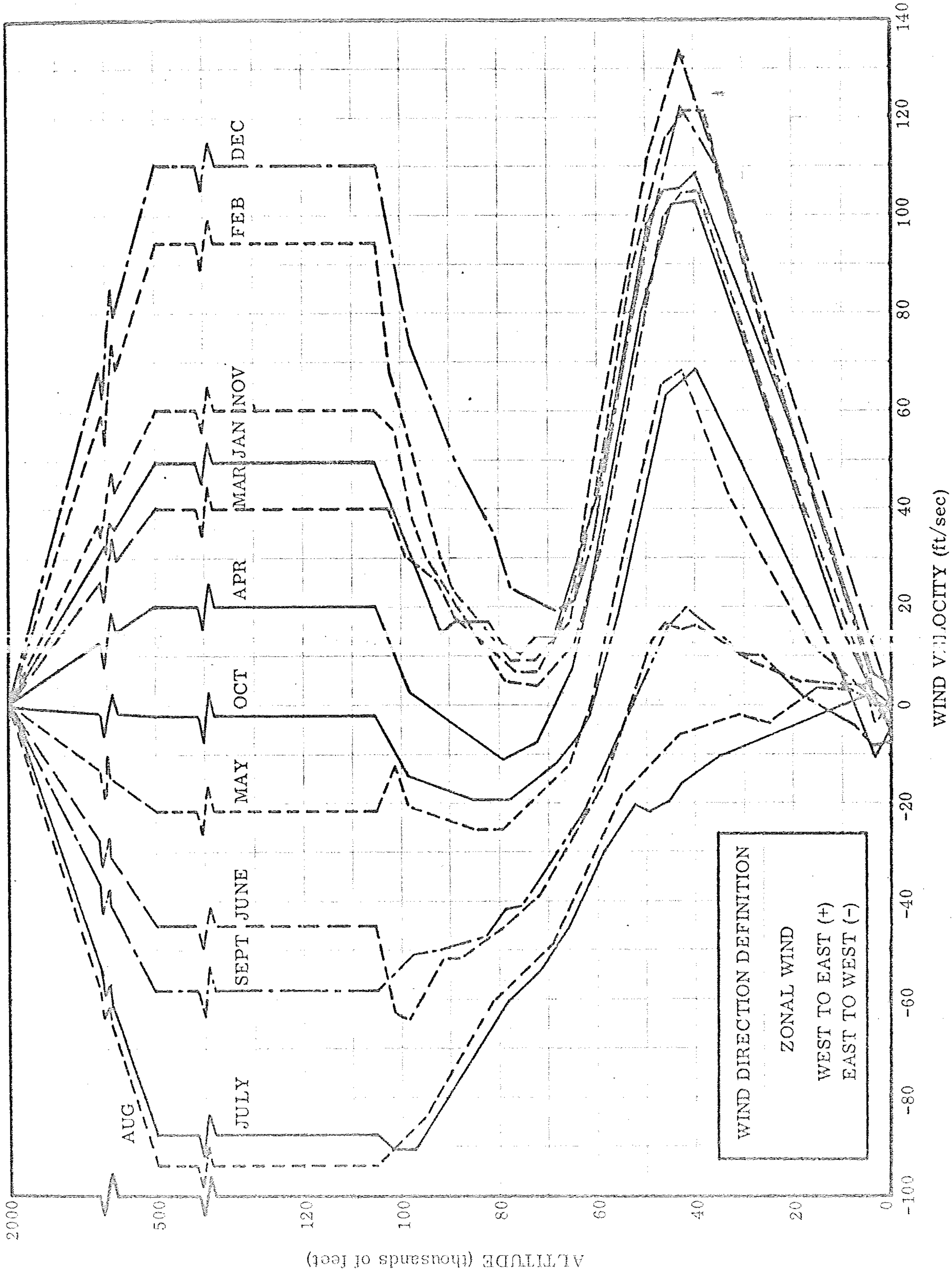


Figure 12-1. Zonal Wind Variations During the Year

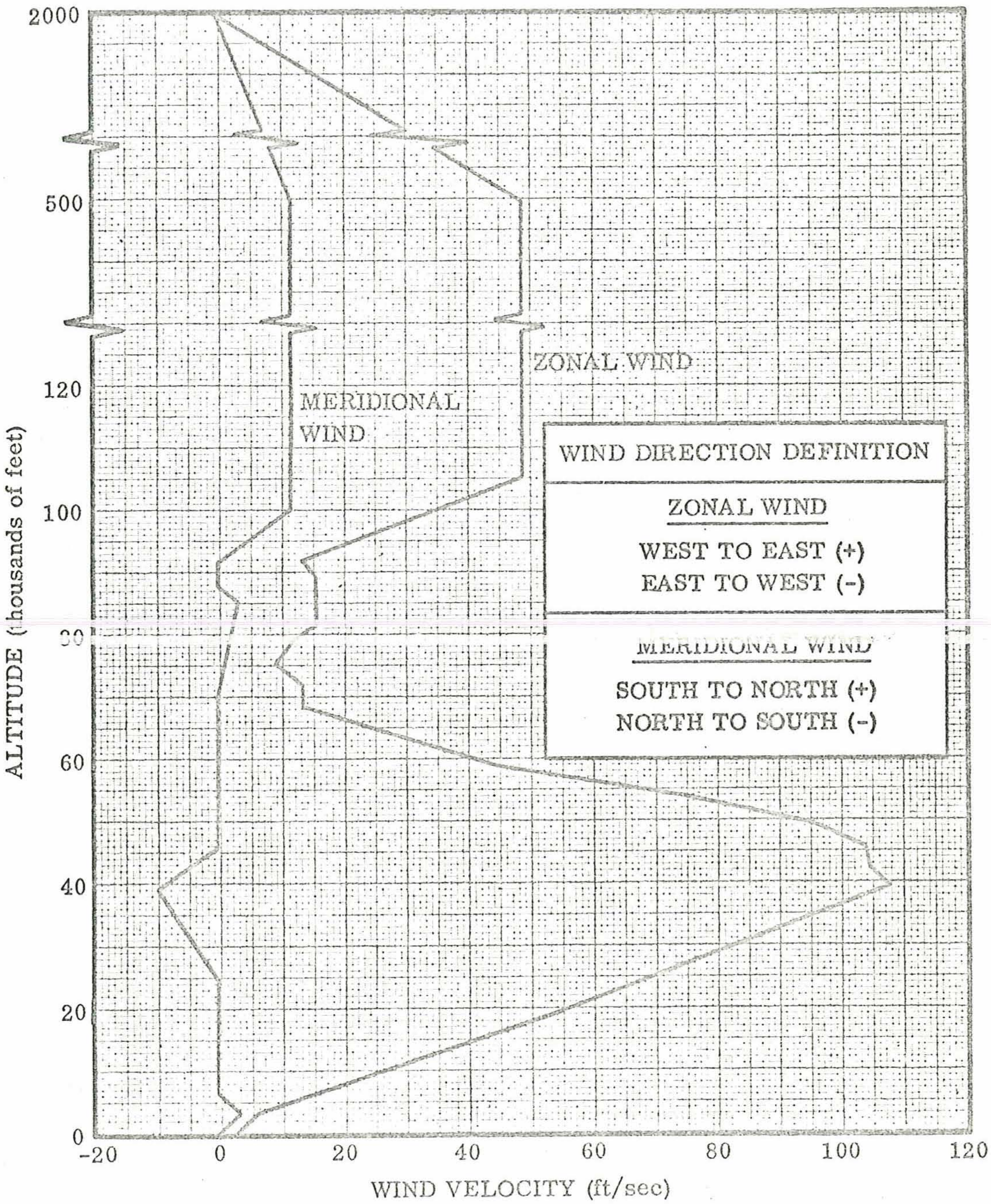


Figure 12-2. January ETR Wind Profile

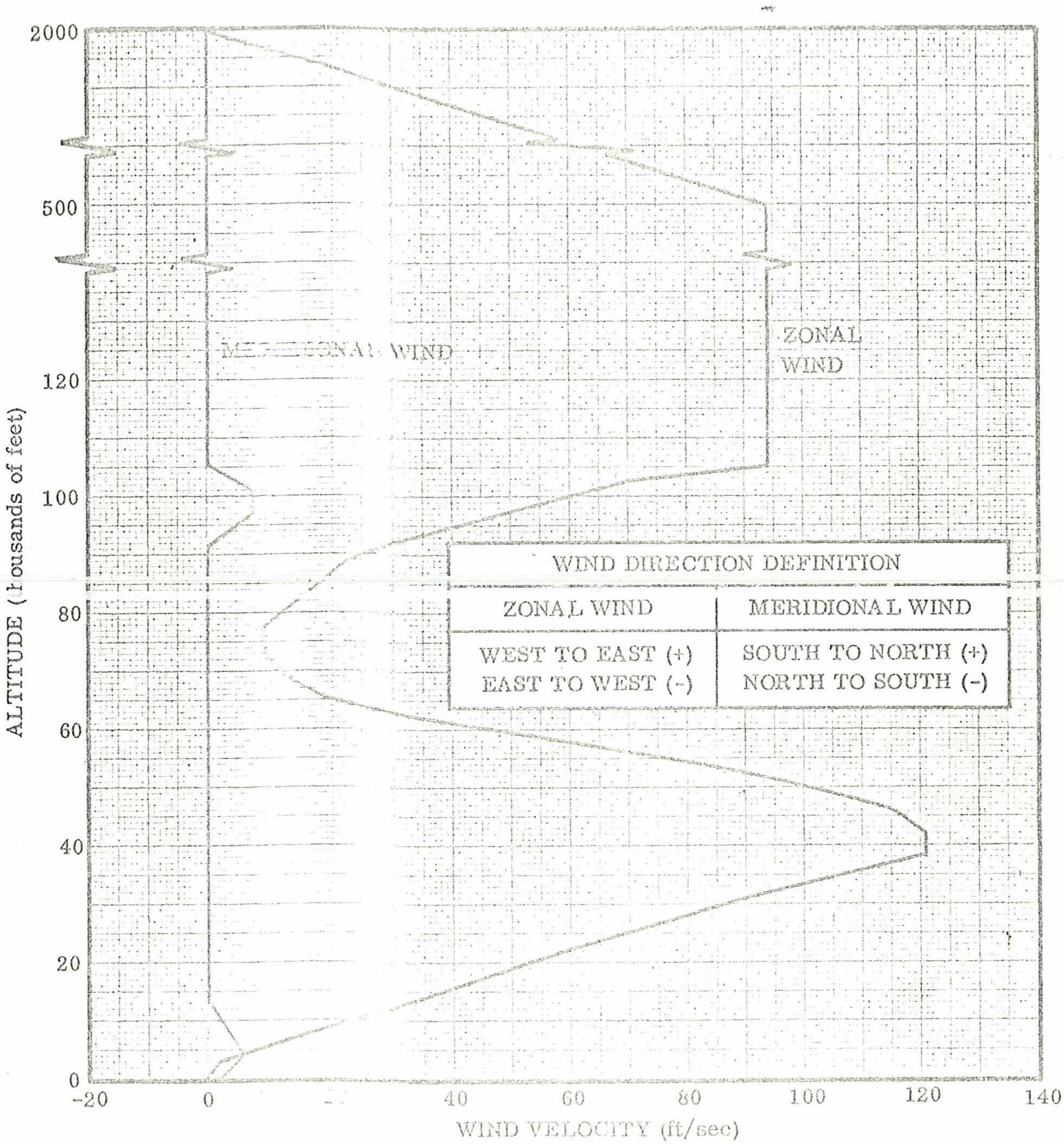


Figure 12-3. February ETR Wind Profile

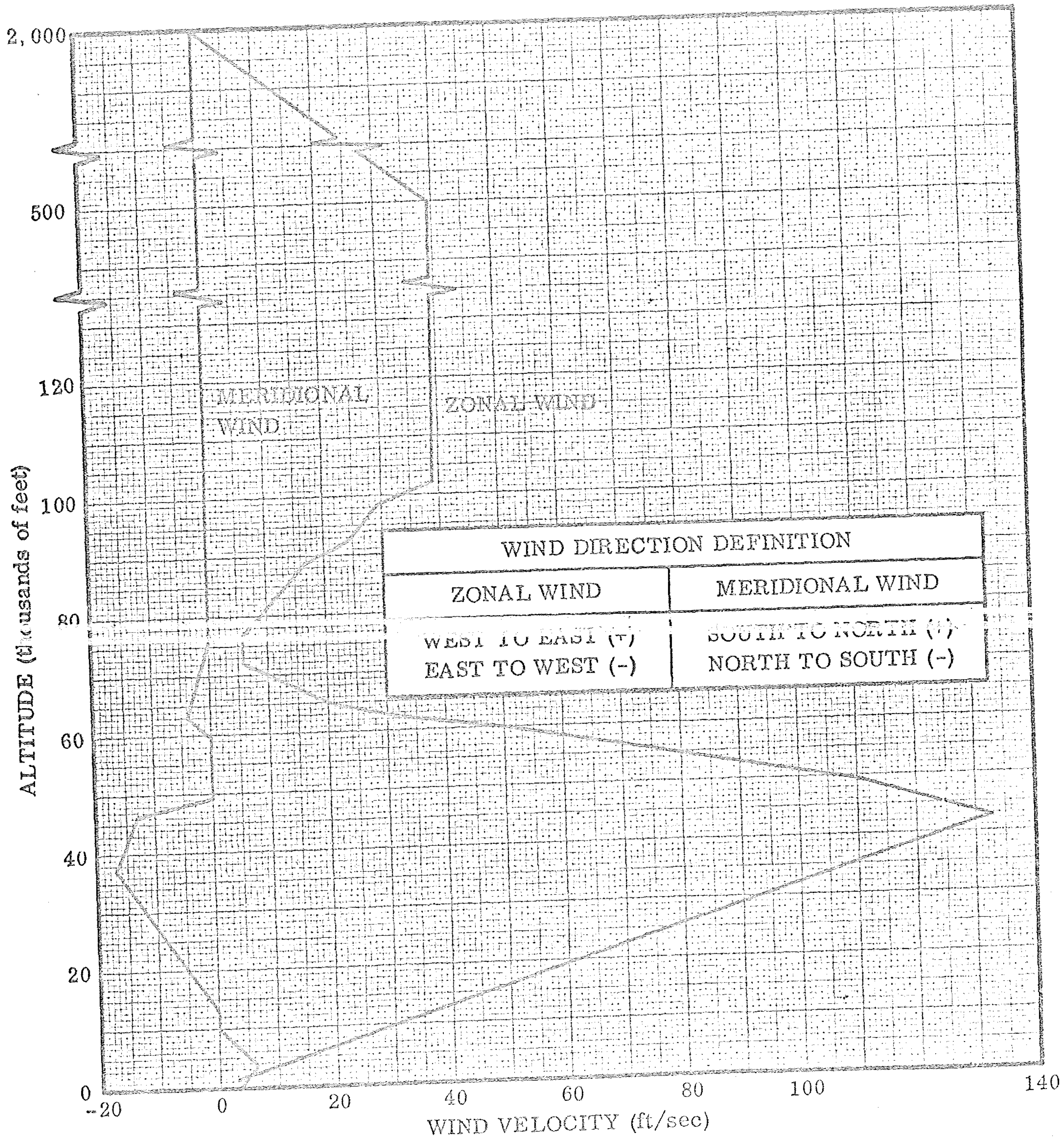


Figure 12-4. March ETR Wind Profile

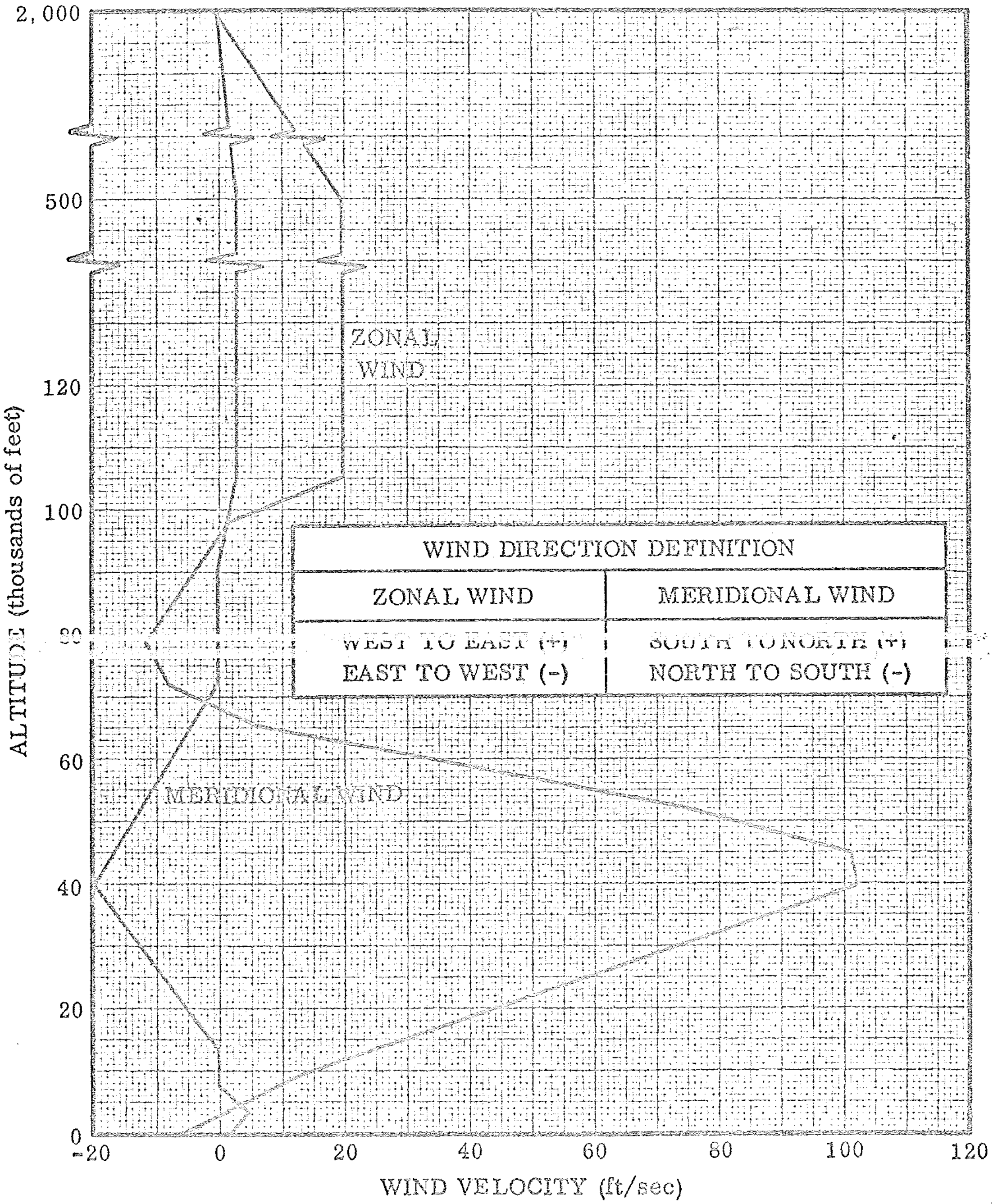


Figure 12-5. April ETR Wind Profile

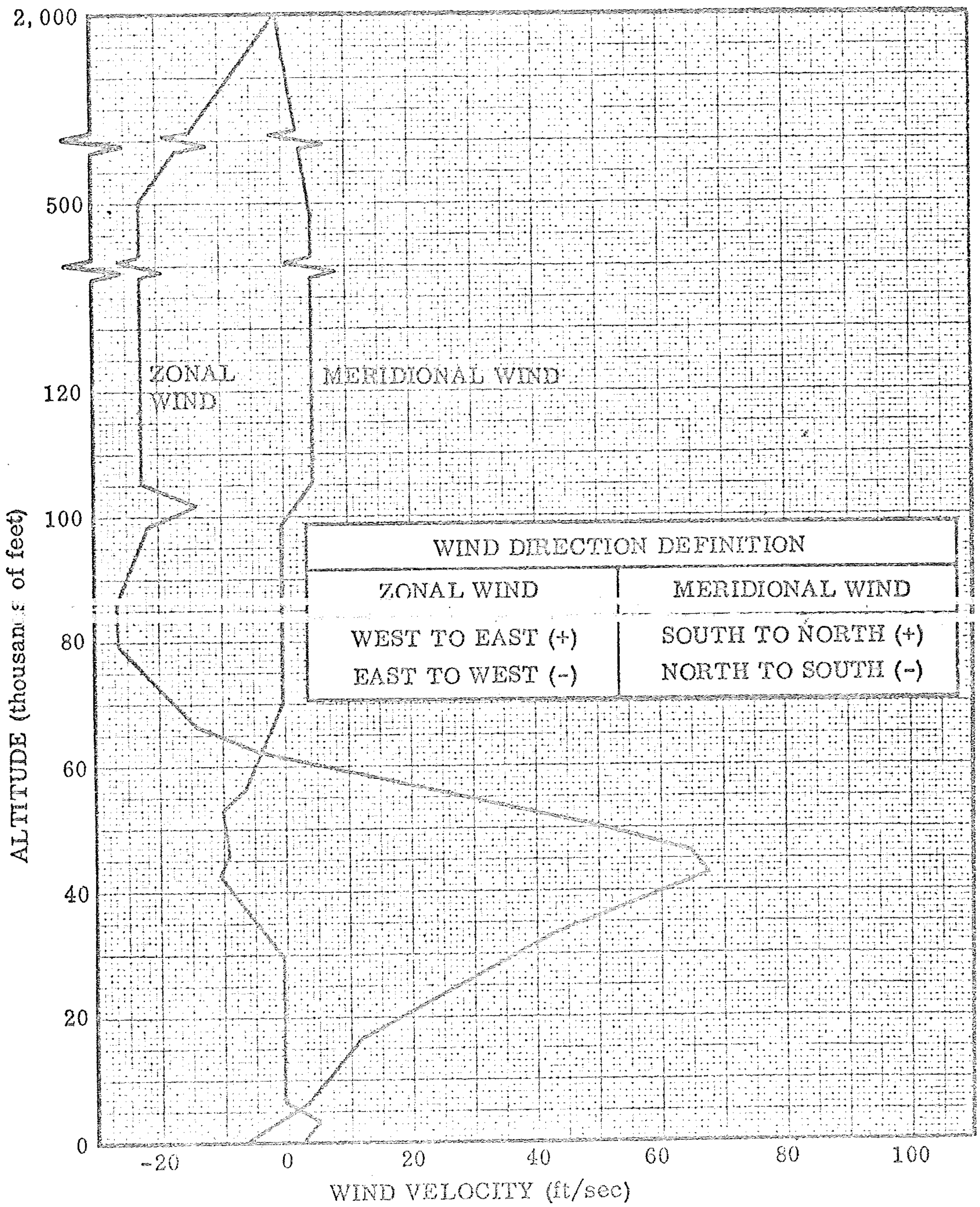


Figure 12-6. May ETR Wind Profile

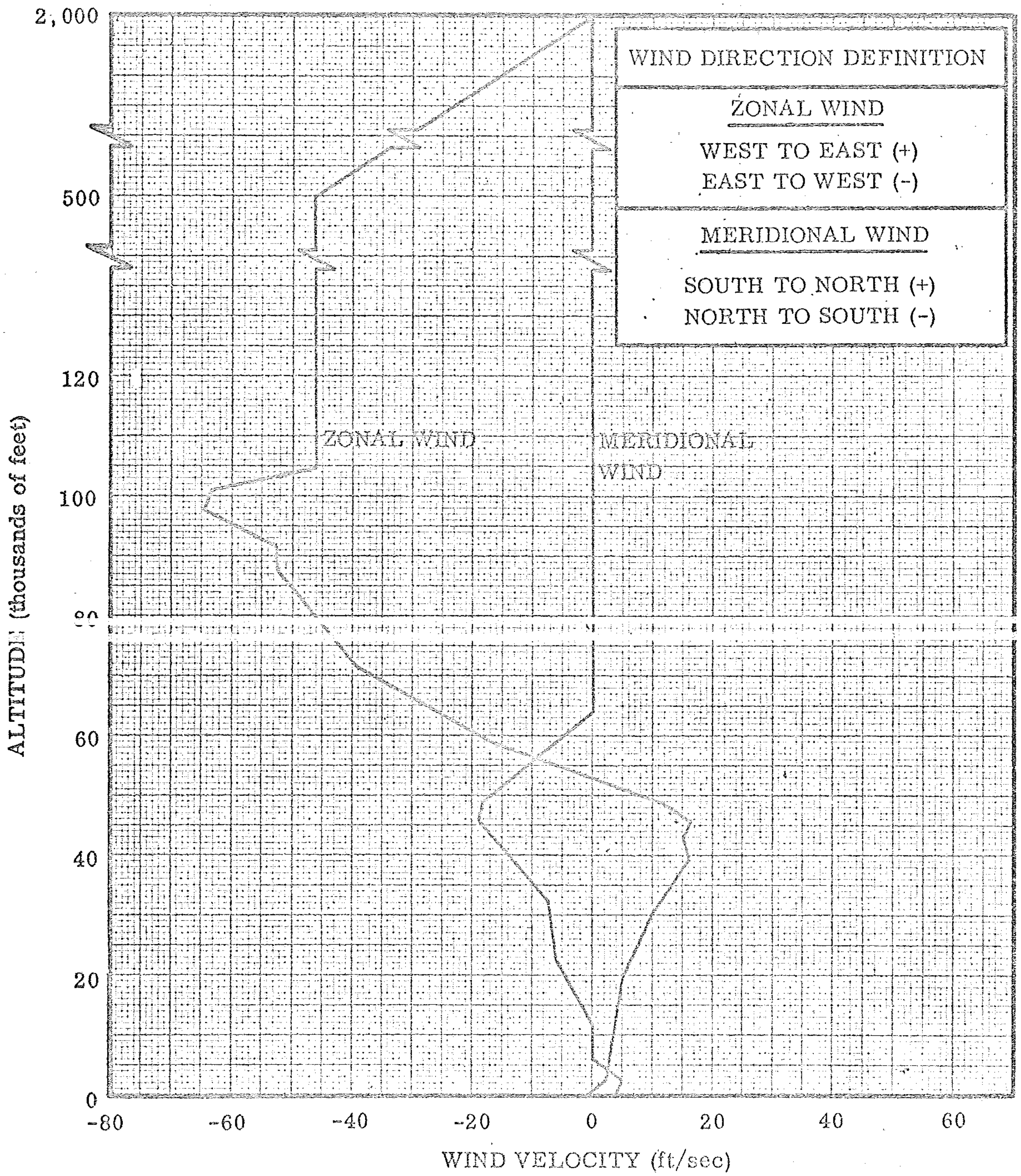


Figure 12-7. June ETR Wind Profile

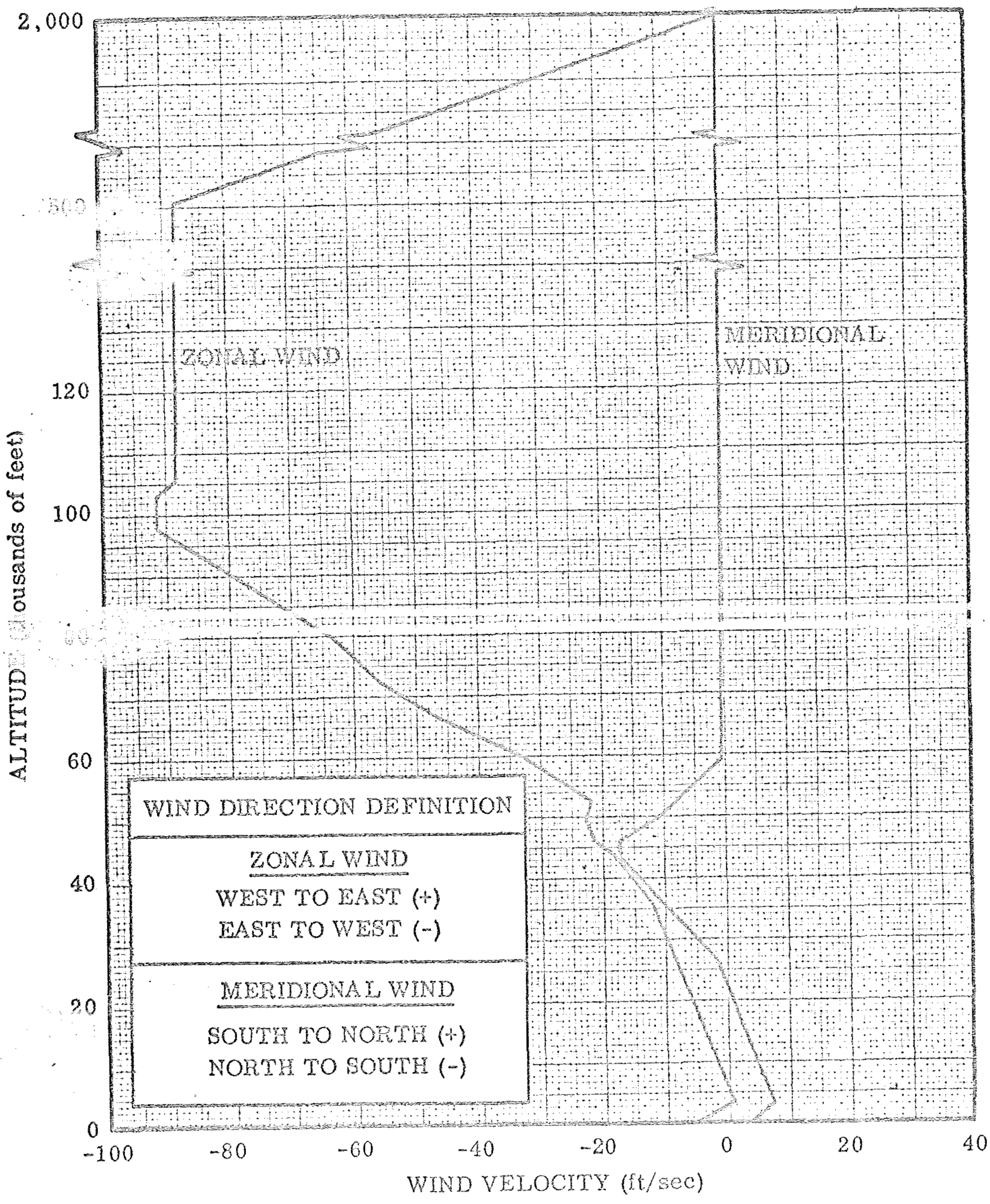


Figure 12-8. July ETR Wind Profile

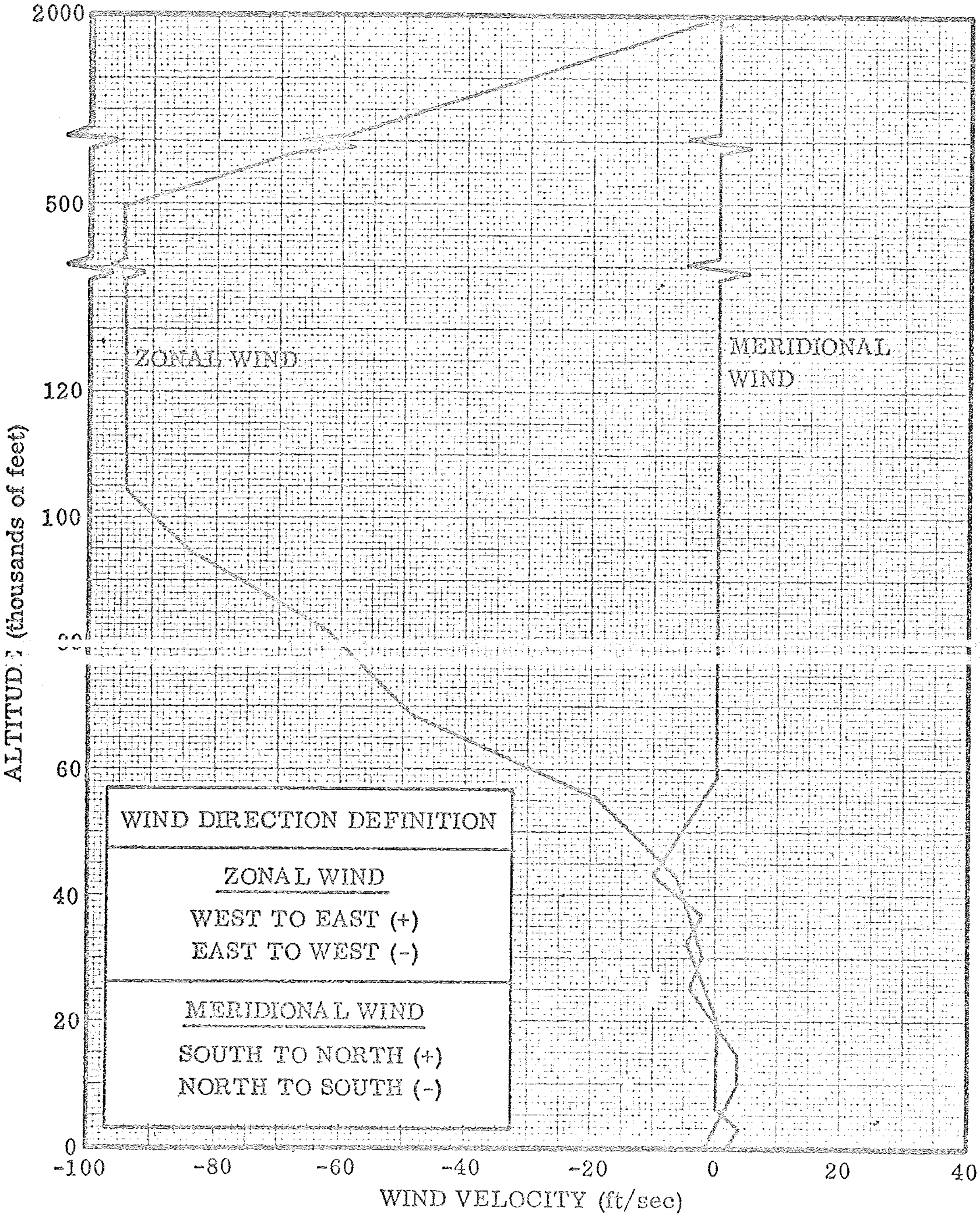


Figure 12-9. August ETR Wind Profile

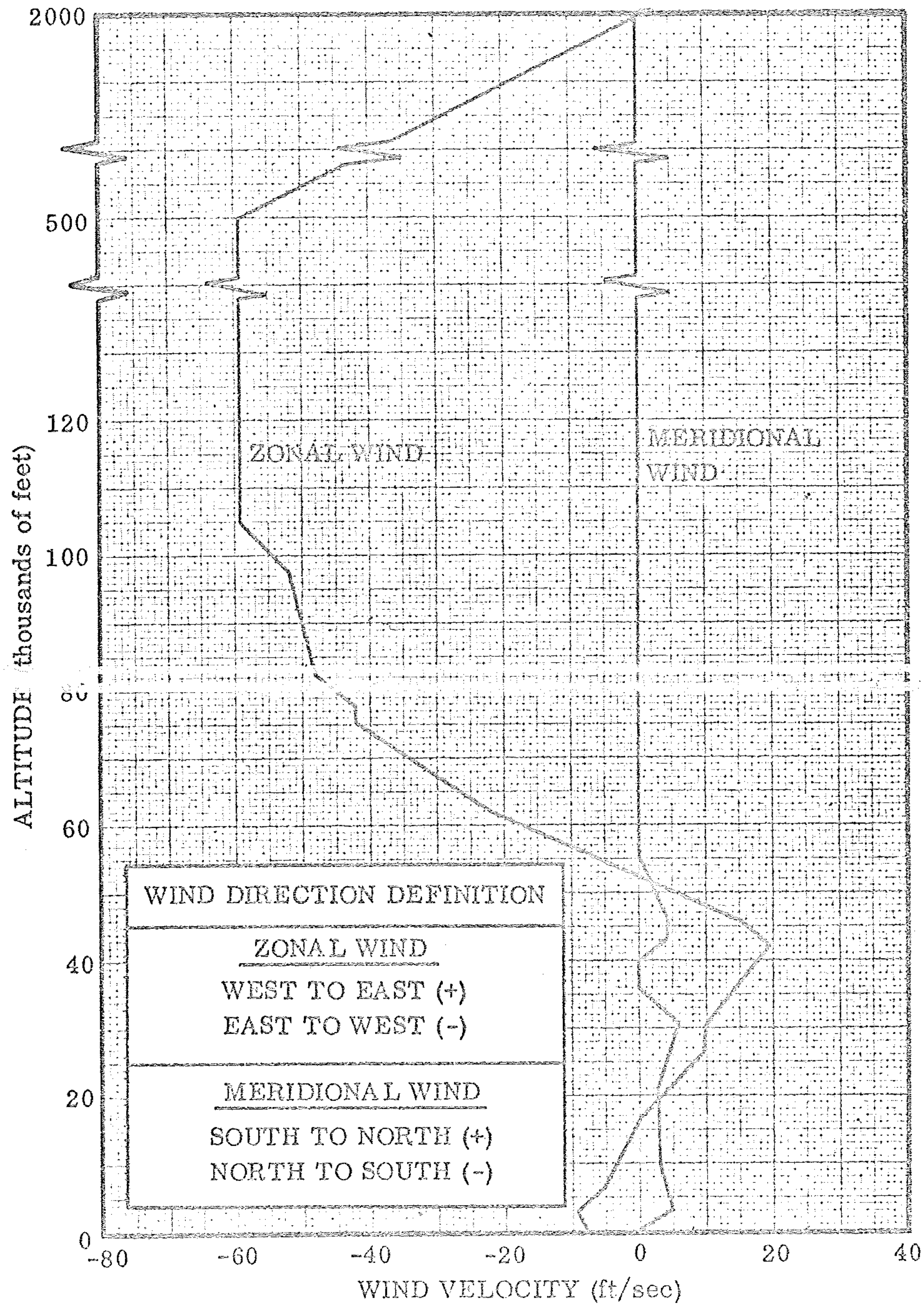


Figure 12-10. September ETR Wind Profile

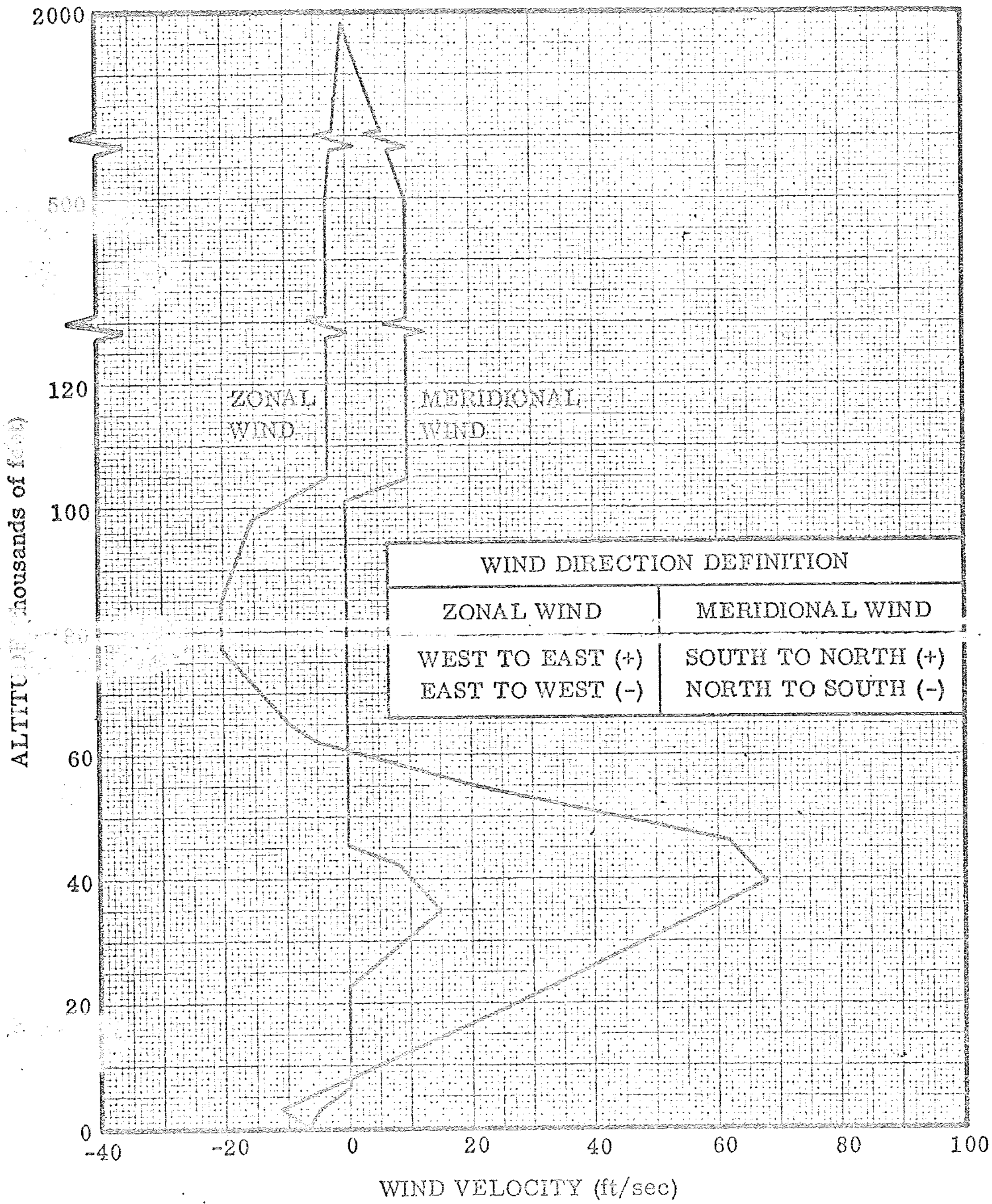


Figure 12-11. October ETR Wind Profile

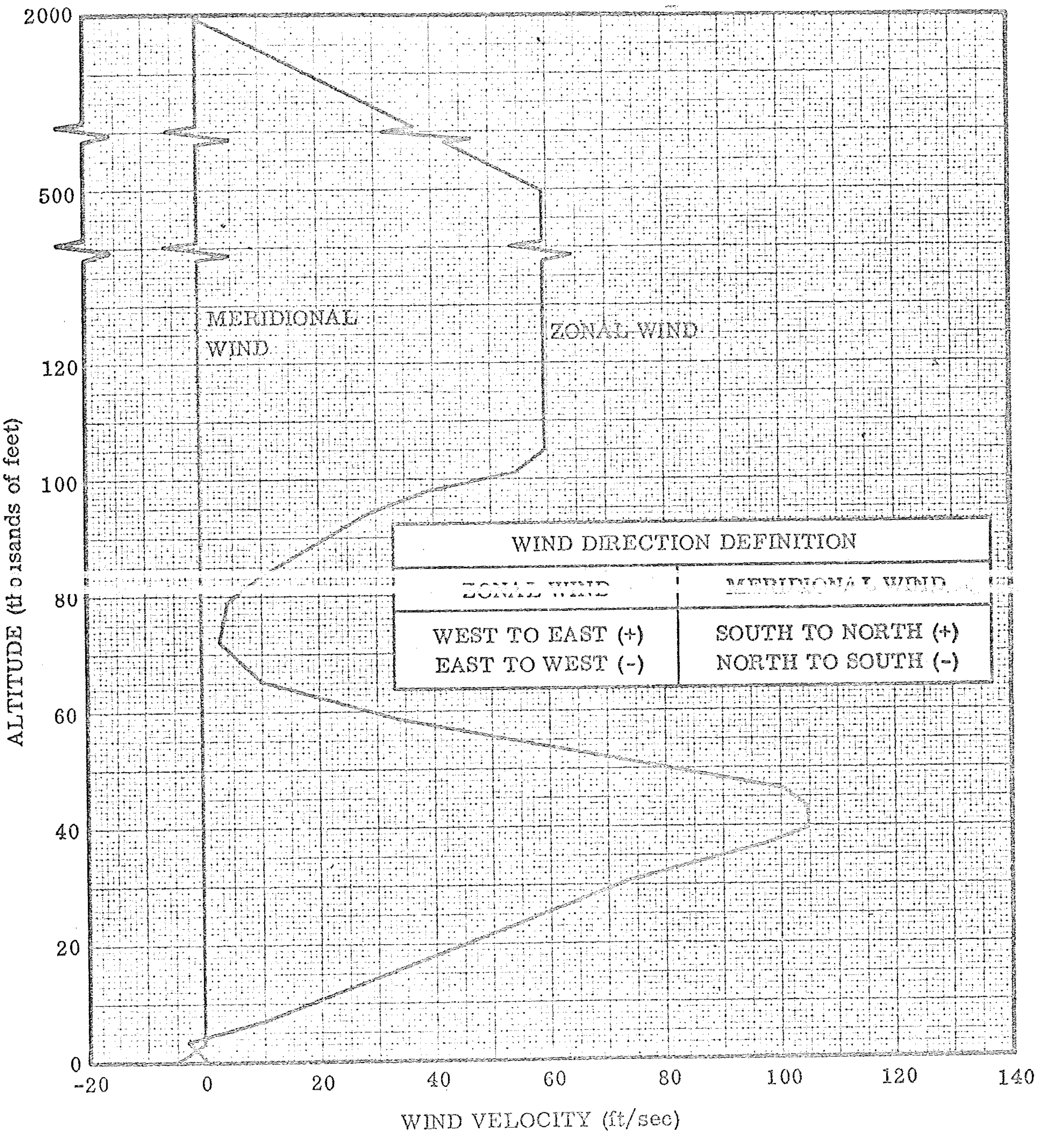


Figure 12-12. November ETR Wind Profile

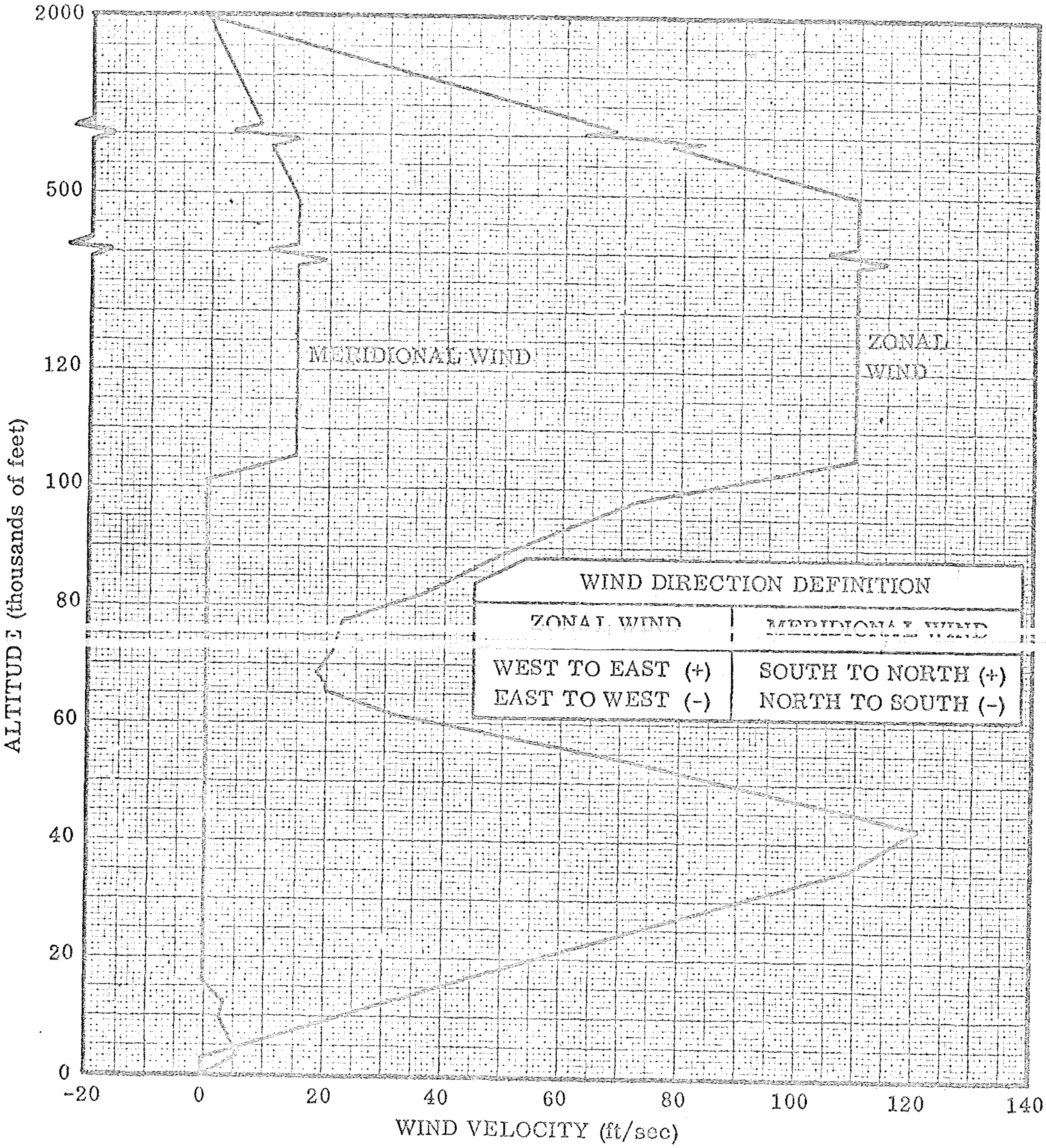


Figure 12-13. December ETR Wind Profile

12.2 MOON DATA

12.2.1 MOON POTENTIAL FUNCTION

The potential function of the moon assumed, Reference 21, is

$$\phi(R, \mu, \lambda) = \frac{GM_m}{R}\left[1 + J\left(\frac{R_m}{R}\right)^2 \left(\frac{1}{3} - \sin^2\mu\right) + L\left(\frac{R_m}{R}\right)^2 \cos^2\mu \cos 2\lambda\right]$$

where,

$GM_m = 4.9027779 \times 10^3 \text{ km}^3/\text{sec}^2$

R_m = Mean lunar radius = 1738.09 km

R = Selenocentric radius

μ = Selenographic latitude

λ = Selenographic longitude

$J = 3106.6 \times 10^{-7}$

$L = 621.48 \times 10^{-7}$

12.2.2 MOON ROTATION RATE

The moon rotation rate (ω_M) assumed, Reference 21, is given by

$$\omega_M = \frac{360}{2360591.545+0.014T} \text{ deg/sec}$$

where T is the number of Julian centuries of 36,525 days from 1900 January 0.5 U.T. (Julian date = 2,415,020.0).

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APPENDIX A

The purpose of Appendix A is to set forth a list of conversion factors currently in use for trajectory computations. These conversion factors, obtained from Reference 21, are presented in Table A-1.

Table A-1. Conversion Factors

FROM	TO	MULTIPLY BY
Inches	Centimeters	2.54
Feet	Meters	0.3048 (Exact)
Statute Miles	Kilometers	1.61
Nautical Miles	Kilometers	1.852
Nautical Miles	Feet	6076.1155
$Meters^2/Second^2$	$Feet^2/Second^2$	10.7639103838